AEC SNSO 3063-1

EXECUTIVE SUMMARY

FINAL REPORT- PHASE I, SENSE STUDY

STANDARDIZATION AND ECONOMICS OF NUCLEAR SPACECRAFT

PREPARED BY:

FAIRCHILD
INDUSTRIES
Germantown, Maryland 20767

FOR:

SPACE NUCLEAR SYSTEMS DIVISION
UNITED STATES ATOMIC ENERGY COMMISSION
AEC CONTRACT AT (49-15)-3063

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ABSTRACT

Feasibility and cost benefits of nuclear-powered standardized spacecraft are investigated. The study indicates that two shuttle-launched nuclear-powered spacecraft should be able to serve the majority of unmanned NASA missions anticipated for the 1980's. The standard spacecraft include structure, thermal control, power, attitude control, some propulsion capability and tracking, telemetry, and command subsystems. One spacecraft design, powered by the radioisotope thermoelectric generator, can serve missions requiring up to 450 watts. The other spacecraft design, powered by similar nuclear heat sources in a Brayton-cycle generator, can serve missions requiring up to 2200 watts. Design concepts and trade-offs are discussed. The conceptual designs selected are presented and successfully tested against a variety of missions. The thermal design is such that both spacecraft are capable of operating in any earth orbit and any orientation without modification. Three-axis stabilization is included. Several spacecraft can be stacked in the shuttle payload compartment for multi-mission launches. A reactor-powered thermoelectric generator system, operating at an electric power level of 5000 watts, is briefly studied for applicability to two test missions of diverse requirements. A cost analysis indicates that use of the two standardized spacecraft offers sizable savings in comparison with specially-designed solar-powered spacecraft. The savings are in addition to those to be realized by the component and subsystem standardization program.

LIST OF FIGURES

LIST OF TABLES

1 INTRODUCTION

This report presents the results of a three-month study and preliminary design effort on standardized nuclear-powered spacecraft. The study was performed for the AEC/NASA Space Nuclear Systems Division in support of the current NASA program aimed at reducing the cost of space missions.

In its role as the future space booster, the Space Shuttle, through standardization, vehicle re-use, and large payload capacity, will significantly reduce the cost of delivering payload to orbit. This has a strong effect on overall cost structure and provides an incentive to extend the savings to payload systems. Much of the NASA effort is concentrated on the definition of standard components and subsystems which can be incorporated into future spacecraft without redevelopment and requalification.

However, even with standard components, a major portion of development, test, and engineering (DTE) costs will remain if the spacecraft itself, its structure, electrical power system, thermal control and vehicle dynamics continue to be highly mission- and orbit dependent, as at present. This study investigates whether the use of nuclear power in the era of the Space Shuttle would permit standardization of the spacecraft as a whole. The motivation behind such standardization is the effort to eliminate, or sharply reduce, the costs associated with developing a new, specially designed, spacecraft for each mission. The DTE costs have historically accounted for approximately 70% of the overall cost of space programs.

These factors suggest the need for a thoroughly new approach to spacecraft design, and a reevaluation of the commonly accepted basis for selecting the spacecraft electric power systems. Hitherto, weight and volume considerations have been paramount and have usually dictated that spacecraft be tailor-made to fit each mission. In these circumstances the power system with the lowest weight-per-watt or overall cost-per-watt is the logical choice. In most cases this has, quite properly, resulted in the selection of solar-array power systems for auxiliary power. Since the spacecraft is being specially designed to begin with, the special requirements of the solar-array system for the particular orbital conditions of the mission can usually be satisfied in the design.

The unique properties of the Space Shuttle, however, imply that changes in the groundrules are necessary. The large lifting capacity and the substantially reduced cost of placing a pound of payload in orbit will obviate much of the incentive for specially-designed payload equipment to perform each mission. In addition, the mandated use of the Shuttle as a space ferry for all missions will strongly suggest the development of "standard spacecraft" with standard integration modes and procedures for use with this standard booster. These standard spacecraft should be capable of performing a variety of different missions without redesign or requalification. They would simply be outfitted with the specific equipment required for each mission. In concept, this equipment would only need to be integrated with the standard "housekeeping" systems provided by the standard spacecraft, such as telemetry, attitude control, and electrical power generation.

Under these groundrules, a nuclear system may be the more cost-effective choice for auxiliary power for most missions, even if the recurring cost of the generator itself were to remain relatively higher than that of a solar-array system of comparable capability. The fundamental advantage of the nuclear system in this context is that it provides power (and heat) which is not dependent on the sun-angle or occultation period, and is not degraded by trapped Van Allen or solar radiation. These highly orbit-dependent factors generally require solar-array systems to be specially designed for each mission. The use of solar arrays, in turn, tends to require the development of a custom-designed spacecraft for each mission, despite the relaxation of the weight and volume limitations permitted by the shuttle.

Nuclear systems, by contrast, are largely insensitive to orbital and environmental factors. For example, the same basic nuclear generator (MHW-RTG) is planned to be used in earth orbit (LES 8/9) and for deep-space missions (Mariner/Jupiter-Saturn). Moreover, the availability of solar-independent waste heat from the nuclear systems should permit the design of a thermal-control system which is largely orbit-and mission-independent, and which permits the adaptation of various types of mission equipment without redesign of the standard spacecraft. This waste heat can be used to compensate for variations in solar input, earth reflection, and load power consumption, thus

maintaining equipment temperature within acceptable limits for different orbits and mission-power profiles.

It is not suggested here that solar-array power could not continue to be used for Shuttle-delivered missions; only that nuclear power more readily lends itself to the design of multi-purpose spacecraft, which would thereby result in significant cost benefits to such missions.

This study was focused on determining the feasibility of designing such a multiple-purpose spacecraft or family of spacecraft, examining its applicability to a wide variety of missions, and estimating the potential cost benefits.

2 CONCLUSIONS

The following are the fundamental conclusions derived in the course of the study:

- Analysis of the most recent NASA mission model for the post-1979 time period revealed that many different missions can be divided into just a few groupings of common power, weight, and size.

- Two standard radioisotope-powered spacecraft, using present-day technology, can satisfy 80 - 90 percent of the unmanned, earth-orbit missions listed in the NASA mission model for the 1979 - 1990 period. The spacecraft provide structure, thermal control, electrical power, attitude and velocity control (AVCS), and tracking, telemetry, and command (TTC) subsystems. The only missions excepted are those few which exceed either the weight-carrying capacity or the attitude control accuracy of the spacecraft as currently designed.

- Two nuclear power system designs, one for each spacecraft, can satisfy all the missions served. The systems are the Radioisotope Thermoelectric Generator (RTG) and the Radioisotope Brayton (RIB) system. For each spacecraft mission, power is provided as required in modular steps, either by addition of complete RTG's to provide power up to 450 watts(e) or by adding fuel capsules to the Brayton system to cover the range from 500 to 2200 watts(e). Both systems can use the same standardized heat source.

• Waste heat from the nuclear generators can be used to maintain spacecraft equipment temperatures within acceptable limits for all altitudes, sun angles, and equipment duty cycles. This is the key factor which permits standardization of the entire spacecraft despite the wide variety of mission conditions encountered.

• Although DOD missions were not included in the study, a separate review indicates that approximately 80 percent of these missions can be served by one or the other of the standard spacecraft designs.

• Savings of 66 percent for four missions using the standardized RTG spacecraft and 46 percent for six missions using the Brayton-spacecraft, in comparison with mission-specialized solar-powered spacecraft, were estimated on the basis of the cost model in the USAF Space Planners Guide. The savings over these missions, over and above those to be derived from the component and subsystem standardization program, are estimated to be 1.7 and 2.8 times the respective development cost of the standardized spacecraft. Thus the development cost can be amortized over relatively few missions and savings thereafter are appreciable. The calculations included both recurring and non-recurring costs, but excluded specialized payload equipment, launch, and operations costs.

• Basic problems arising from the use of radioisotope fuel have been successfully met in previous missions. The routine use of nuclear systems in standard spacecraft should reduce the cost of the specialized ground handling and testing equipment required. A clearer definition of ground operations and safety requirements must await further specification of the shuttle ground time line and abort modes.

• For missions requiring approximately 5 kilowatts(e), the study demonstrated that a standard Reactor Thermoelectric System (RTS) can fit within the shuttle vehicle and be mated to two such diverse missions as a low altitude radar mapping experiment and a synchronous orbit communications system. In this power range, reactor electric power is less costly than solar-array power on a dollar per watt(e) basis. The mission flexibility of the RTS may be expected to provide a further cost advantage in DTE when multiple missions are met.

3 SPACECRAFT DESIGNS

3.1 Functional Requirements and Mission Analysis.

A goal of the study is the conceptual design of nuclear-powered spacecraft capable of satisfying many different NASA missions without significant change in the standard spacecraft or their subsystems. The missions considered were those listed in the 1971 and 1972 NASA mission models (see Ref. S-1 & S-2), with manned and interplanetary missions excluded by direction. The 1972 mission model is considerably reduced from the 1971 version in numbers of missions and flights. In some cases the missions are not yet well-defined and not much more than mission titles and functions are available. No model can be complete since new missions will undoubtedly appear, and extensions of existing missions will probably be added.

Table S-1 is a list of the functions to be performed by the spacecraft and of the subsystems associated with each function. The mission models were reviewed to classify the demands made upon the various subsystems in order to furnish a basis for common designs. The analysis showed a high degree of commonality in functional demands, e.g. the greatest number of spacecraft are earth oriented, with a small number of observatories pointed toward the sun or other celestial objects. Three axis stabilization is required or desirable in most cases. Pointing accuracy of 0.1 degree satisfies the needs of most missions. Higher accuracy is ordinarily required only in knowledge of the orientation, and can be provided by the addition of appropriate sensing equipment and of ground data reduction as required. Telemetry, Tracking, and Command (TTC) requirements involve a fairly low data rate which fits within either VHF or S band and both are compatible with vehicle orientation and with pointing accuracy requirements. High data rate payload communications could also be served if a tracking and data relay satellite system (TDRSS) becomes available. If a few specialized missions with very unusual parameters are removed from consideration, it becomes possible to satisfy all of the remaining missions with a limited number of choices corresponding to the mission parameters.

TABLE S-1 SPACECRAFT FUNCTIONS AND SUBSYSTEMS

FUNCTION	SUBSYSTEM
Physical support of payload and service elements, tie to booster	Structure
Provision of satisfactory environment, disposal of heat	Thermal control, shielding
Provision of electrical power	Power
Propulsion for Stationkeeping, orbit adjustment	Propulsion or AVCS
Orienting spacecraft as required	AVCS (+ TTC if ground command)
Provision of unobstructed view for directional elements	Structure + AVCS
Provision of communications for payload and service elements (limited rate)	TTC
Provide command and control of spacecraft	TTC

These missions were then divided into two groups, depending on whether they required more or less than 500 watts(e). This rather arbitrary division corresponds to the power ranges chosen for the RTG system (up to 450 watts) and for the Brayton system (above 500 watts). Tables S-2 and S-3 summarize some important mission parameters for these groups. The tables show that, with two exceptions, the missions involving the lower power levels also involve spacecraft weighing less than 2000 pounds, while in general the missions requiring more than 500 watts(e) involve weights above 2000 pounds. It was therefore decided that the entire range of requirements could conceivably be met by only two different spacecraft designs. This was subsequently verified by testing the designs against a number of specific missions with differing requirements.

Table S-4 then summarizes a set of basic requirements for standardized spacecraft powered by the two different systems, capable of satisfying the great majority of missions. Missions covered include principally earth observation, communication and navigation, astronomy, and earth and space physics. The excluded missions include the large observatories, whose weight requirements substantially exceed those of all other missions, and certain physics experiments, such as a relativity experiment with extremely precise attitude control requirements. The large observatories could possibly be served by yet another standard spacecraft, but time did not permit such an investigation in this preliminary study.

At higher power levels of five kilowatts(e) or above, which can be supplied by the Reactor Thermoelectric System, the only NASA mission in the 1972 model is a synchronous-orbit high-power communications satellite. For study purposes, a low-orbit side-looking radar mapping mission, which appears possible at approximately 5 kW(e), was added. The application of the same reactor system to these two cases was studied to determine the feasibility of performing two such diverse missions with the same basic power system. By direction, the use of reactor-powered electrical propulsion for orbit transfer was not considered in this phase of the study.

TABLE S-2 MISSION CHARACTERISTICS - LESS THAN 500 WATTS REQUIRED

(1971 and 1972 Mission Models)

PURPOSE	LOW ORBITS					SYNCHRONOUS ORBITS			
	No. of missions	No. of flights	Elec. power watts	Nominal s/c Wt. lbs.	Orbit N. Mi.	No. of missions	No. of flights	Elec. power W	Nominal s/c Wt. lbs.
Earth Observ.	2	15	200	1000	700	5	35	300/400	1000
	1	12	250	1400	900	1	6	400	3000
Comm. Nav.	3	12	420	1000	Ellipt.	1	24	150	500
						3	34	200/400	1000
						1	10	430	1000
Physics	1	7	200	600	400 P				
	3	26	400	1000	Ellipt.				
Astronomy	1	8	40	400	300	1	8	40	400
						1	9	100	1000
	1	15	100	1000	300	1	2	200	2000
	1	1	300	2000	300	1	1	230	3000
TOTALS	13	96				15	129		

TABLE S-3 MISSION CHARACTERISTICS - OVER 500 WATTS REQUIRED

(1971 and 1972 Mission Models)

PURPOSE	LOW ORBITS No. of missions	No. of flights	Elec. power watts	Nominal s/c Wt. lbs.	Orbit N. Mi.	SYNCHRONOUS ORBITS No. of missions	No. of flights	Elec. power W	Nominal s/c Wt. lbs.
Earth Observ.	3	40	600	2500	500 P	1	5	1500	2500
	2	9	1200	2500	500 P				
Comm./Nav.	1	12	500	600	Ellipt.	1	12	500	600
						1	11	580	1500
						4	33	700/1000	2000
						2	14	600/1000	3000
						2	23	900/2000	3500
TOTALS	6	77				11	98		

TABLE S-4 STANDARDIZED SPACECRAFT DESIGN GOALS

		RTG S/C	BRAYTON S/C
1.	Power:	Up to 450 W(e) in modular steps	500 to 2200 W(e) in modular steps
2.	Weight:	1000 to 2000 lbs	up to 4000 lbs

RTG and BRAYTON S/C

3. Ferrying: Stacking capability within shuttle, Agena/Centaur or chemical tug interface as necessary.
4. Propulsion: Small orbit change or stationkeeping capability.
5. Power: 28 Volt DC, ± 2 % regulation
6. Attitude Control: Three-axis stabilization, 0.1 degree accuracy
7. TT & C: Standardized, 1 MBS data rate
8. Orbit: Low orbit, any inclination, up to synchronous orbit
9. On-orbit life: 3 year minimum
10. Thermal environment: Equipment thermal environment between -10° C and +40° C for all orbits, spacecraft orientations, sun occultations, and spacecraft power levels.
11. Pointing: Any direction, including earth-oriented

3.2 Nuclear Electric Power Systems and Launch Vehicles. Radioisotope Thermoelectric Generators (RTG) producing 40 watts(e) at the beginning of life (BOL) have been flight qualified and have operated successfully on previous programs. Such units are represented by the SNAP-19/Pioneer generator shown in cutaway view in Figure S-1. The larger Multi-Hundred Watt (MHW) generator shown in Figure S-2 is expected to be operational by the end of 1974. This unit will provide approximately 150 watts BOL, and is the basic RTG power source considered in this study. The MHW unit can be fueled either in a late stage of ground operations, or in orbit, if preferable, by use of the remote manipulators in the shuttle. Precedent for post-launch fueling exists in the Apollo program, wherein an RTG was fueled on the lunar surface. Table S-5 presents some characteristics of the SNAP-19 and MHW units.

Figure S-3 is a schematic of the Radioisotope Brayton System (RIB), which has been selected as the most promising for the 0.5 to 2 kilowatt(e) level. It is fueled by the same heat sources as the MHW-RTG, but uses a turbo-alternator driven by an inert gas working fluid and supported on gas bearings, in the power conversion system. By adjusting only the pressure of the working fluid, the same generator hardware set can be fueled with either one, two, or three fuel capsules, thus increasing output power over a large range with relatively little change in efficiency. Figure S-4 is a photograph of a Brayton model with a single heat source in one possible configuration, and Table S-6 presents some characteristics of the system for a given fuel loading. The system is designed to operate at constant speed of rotation and power level; this is accomplished by application of a parasitic load to offset variations in power demand.

Figure S-5 is a schematic of the Reactor Thermoelectric System, whose general configuration is shown in Figure S-6 and parameters listed in Table S-7. This system can handle the power range from 3 to 10 kW(e). Figure S-7 shows the thermoelectric converters used in the RTS.

TABLE S-5

REFERENCE RTG PARAMETERS *

Parameters	SNAP-19/ Pioneer	MHW
Nominal Max. Output Power, watts (e):		
BOL	40	150
3 years	32 design life	135
5 years	--	130
12 years	--	115 design life
Design Voltage at Max. Power, volts	4.2	30
Initial Heat Source Loading, watts	650	2400
Active Radioisotope (PU-238) half life, years	86.4	86.4
On-Pad Power Available, watts (e)	40	50-90
Generator; Length, inch	11	21
; Envelope diameter, inch	20	16
; Fin Height, inch	6.75	2
Case Material	Mg-Th	Be
Heat Source; Length, inch	6.5	16.60
OD, inch	3.5	7.25
Weight, lb.	30	85
Thermoelectric Couples, no.	90	312
, series x parallel array	45 x 2	156 x 2
Nominal BOL Temperatures, ^{o}F		
; radiator fin-root	330	550
; TE hot junction	975	1830
; heat source exterior	1050	1940
; fuel centerline	1815	2600
Nuclear Radiation at 1 meter; R/hr	0.3	
; neuts/cm^2 sec.	3×10^3	
Magnetic Field at 1 meter, gammas	40-70	(No data available)

*Data furnished by United States Atomic Energy Commission

TABLE S-6

REFERENCE BRAYTON PARAMETERS**

Parameters	No. of Heat Sources 1	2	3
Net Conditioned Output Power, watts (e); BOL	689	1588	2273
Conversion Efficiency, %; BOL	28.7	33.1	31.6
Design Output Voltage, volts DC		120*	
Gas Flow Rate (approximate), lb/sec.	0.1	0.2	0.3
BRU Rotation Rate, RPM	50,000		
Gas Temperatures (nominal), ^{o}F;			
; radiator inlet		300	
; radiator outlet		70 *	
; compressor outlet		250	
; recup. outlet		1250	
; turbine inlet		1600	
Heat Source Skin Temp. (max.), ^{o}F		1800	
Weights, lb. ; BRU		25.3	
; recuperator		68.4	
; radiator		161.6*	
; ducting		11.7	
; HSHX		171.0	
; superinsulation		78.0*	
; structure		55.0*	
; electronics		15.0	
; parasitic load		20.0	
; heat sources	45	90	135
TOTAL	651	696	741

* These values may not apply to the specific designs shown elsewhere in this report.

** Data furnished by NASA Lewis Research Center

TABLE S-7

REFERENCE REACTOR/TE PARAMETERS * *

Parameters	5 kWe System		10 kWe System	
	BOL	EOM#	BOL	EOM#
Electrical Power at 30 VDC, KW				
Gross	5.13	5.13	10.3	10.3
Net at Mating Plane	5.0		10.0	
Reactor Thermal Power, KWT	86.2	93.0	198	213
Reactor Outlet Temperature, ^{o}F	1140	1200	1142	1200
Flow Rate, Lbs/Sec				
Primary Loop	4.71	4.67	10.8	10.7
Secondary Loop	2.85	2.84	6.7	6.6
Pressure Drop, PSI				
Primary Loop	0.84	0.84	1.07	1.04
Secondary Loop	1.25	1.25	1.7	1.6
Converter Efficiency, %	6.83	6.29	6.1	5.6
Radiator Area, Ft^2	246		522	
Base Diameter, Ft*	6		9	
System Overall Height, Ft*	22.3		31.5	
Radiator Cone Half-Angle, ^{o}*	8		8.5	
Number of TE Modules				
Power	16		36	
Pump	3		6	
Total Weight, Lbs	1785		2940	
Distance from Base Diam. to CG, Ft*	15.1		20.6	
Temperatures, ^{o}F, Primary Loop				
Reactor Outlet	1140	1200		
Power Module T	82	28		

Based on five-year lifetime.

* These values may not apply to the specific designs shown elsewhere in this document.

** Data furnished by United States Atomic Energy Commission

TABLE S-7 (Continued)

REFERENCE REACTOR/TE PARAMETERS

Parameters	5 kWe System		10 kWe System	
	BOL	EOM#	BOL	EOM#
Temperatures, ^{o}F; Primary Loop (Continued)				
Pump Module	98	102		
Reactor Inlet	1060	1110		
Reactor Δ T	80	90		
Temperatures, ^{o}F; Secondary Loop				
Radiator Outlet	459	471		
Power Module Δ T	124	134		
Pump Module Δ T	145	156		
Radiator Inlet Δ T	584	607		
Radiator	125	136		
Temperatures, ^{o}F; Radiator Fin Root				
Inlet	582	605	(Data not furnished)	
Outlet	457	469		
Total Radiation Dose at Base Plane (20 ft), NVT		10^{12}		
RAD		10^{6}		
Weight; Lbs:				
Reactor	482			
Shield, Gamma	138			
Shield, Neutron and Case	223			
Converters	192			
EM Pumps	77			
Volume Accumulators (3)	38			
Piping	127			
Radiator/Structure	452			
Electric Wiring	46			
Miscellaneous	10			
Total	1785		2940	

Figure S-8 is a sketch of the shuttle orbiter vehicle, showing its payload compartment area. Figure S-9 is a sketch of its remote manipulator system which can be used to handle payloads. The shuttle system can boost up to 65,000 pounds into a 100 nautical mile circular orbit (28.5° inclination) 40,000 pounds into a polar 100 n.mi. orbit, or 50,000 pounds into a 270 n.mi. orbit at 28.5°. For normal boost periods, adequate heat storage is readily available in the shuttle to handle the waste heat from the radioisotope sources in the RTG and Brayton system. The doors of the payload bay are usually opened after leaving the atmosphere, which permits radiating some of the heat to space. Extended storage periods in the shuttle, particularly with bay doors closed, requires auxiliary heat storage or cooling, since the active temperature control system now planned for the shuttle payload compartment cannot handle the heat rate from the radioisotope sources.

3.3 The RTG Bus Design.

A standard RTG-powered spacecraft or "bus" design evolved from the study. It is shown in sketch form in Figure S-10 and layout in Figure S-11. The design appears to be suitable for all the missions listed in Table S-3. Detailed studies confirmed the applicability to two specific missions.

The bus has the form of a hexagonal torus six feet across and three feet high; the space in the center is occupied by the fuel tank and thrusters of a hydrazine monopropellant liquid propulsion system used for orbit adjustment and attitude control. Vehicle weight including payload can range from 1000 to 2000 pounds, depending on payload and power requirements.

Three compartments of the hexagonal prism contain the standard subsystems and three compartments are available for payload. Viewing or communicating equipment requiring a clear field of view is mounted at one end of the prism; for example, in earth-oriented missions, at that end which faces earth. Electronic and other temperature sensitive payload is mounted internally on the panels forming the two ends of the spacecraft. The faces have mounting holes and connectors for flexible payload placement. Large temperature-insensitive payload components, such as antennas, will be mounted external to the spacecraft. Depending on the power level required, one, two, or three RTG units are mounted externally on the side faces of the prism, multiple units symmetrically and a single unit suitably counter-weighted. Access to internal equipment is through those side faces not bearing RTGs.

Standard subsystems include a 28 volt DC regulated ($\pm$ 2%) power supply and a VHF and S band telemetry system with a 10^3 bit/second capacity. The hydrazine propulsion system has fuel capacity for a 1170 foot/second velocity increment; because the thrusters are fixed, large increments will require spinning the vehicle. The configuration is chosen to facilitate balance and mass distribution suitable for stable spinning about the thrust axis. Small velocity increments, such as are applied in stationkeeping, do not require spinning; the attitude control system offsets any momentum due to misalignments of the central thrusters from the centroid of the vehicle. The three axis stabilization system employs reaction wheels and the hydrazine monopropellant thrusters for unloading the wheels, when necessary. Temperature of the hydrazine tank is suitably regulated by electric heaters.

The thermal control system provides a regulated environment between -10°C and $+40^{\circ}$C for all spacecraft internal equipment in all conditions, i.e. regardless of occultations of the sun, aspect of the sun, earth albedo radiation and internally generated heat from spacecraft and mission equipment. This thermal independence of mission conditions makes it possible for the spacecraft to be used for a variety of missions. This is achieved by a thermal control system which uses part of the waste heat from the RTG units to overcome variations in heat originating from all other sources.

The case and fins on the RTG radiate most of the heat produced directly to space; their high temperature makes them effective radiators. Between the surface plate bearing the RTG and the rest of the spacecraft is a louver system which controls the rate at which heat is radiated from this plate to the other parts of the bus. The movable vanes of the louver system automatically regulate the spacecraft temperature against variations due to changing heat inputs from other sources. The temperature of the spacecraft end panels is maintained relatively uniform by the use of heat pipes. Standard emissivity coating and insulation techniques control the payload compartment temperature within the proper range despite differences in equipment power level and solar- and earth-originating radiation.

Figure S-12 shows the effect of variations in power levels and mission conditions on the temperature of the end panels that support the equipment. These curves demonstrate the effectiveness of the system in providing a satisfactory environment in all conditions.

Table S-8 is a weight breakdown by subsystem of the maximum size 2000 pound RTG spacecraft carrying three RTG units. Note that this includes a large amount of propellant (370 lbs) for orbit adjustment purposes, and a battery in the electrical system to provide for periods of peak power exceeding the RTG capacity. Mission equipment weight allowance is 764 pounds.

Figures S-13 and S-14 show two means of mounting the RTG bus within the orbiter vehicle, one mounting directly to the attachment points in the payload compartment, and one mounting via a pallet capable of carrying both RTG and Brayton buses individually or in combination. The pallet provides a versatile mounting structure which avoids placing bending moments on the orbiter vehicle.

Validity of the standard spacecraft concept was tested against the specific requirements of two different missions.

Figure S-15 is a sketch of the bus carrying the payload of the synchronous-orbit Tracking and Data Relay Satellite (TDRS), with the antennas deployed in operating configuration. Figure S-16 is a layout of the same satellite in the stowed configuration used during transportation. Figure S-17 is a sketch of the RTG bus bearing the payload proposed for the TIROS follow-on Earth Meteorological Satellite. The standard spacecraft appears to be readily adaptable to both missions, although an economic analysis (Section 4) indicates that the former (TDRS) is more economically performed with the Brayton spacecraft described below.

3.4 The Brayton Bus Design.

The standard Brayton-powered spacecraft design is shown in sketch form in Figure S-18 and layout form in Figure S-19. The design is suited for the missions requiring from 500 to 2200 watts of electrical power, and was also tested for adaptability to two specific missions.

TABLE S-8

RTG BUS - WEIGHT ESTIMATE

PRIMARY STRUCTURE		
• Shells	84.0	
• Ribs	27.0	
• Rings	24.5	
• Support FTG (4)	12.5	
• Stiffeners	20.0	
• RTG Supt. (3)	6.0	
• Contig (10%)	16.0	(190.0)
EQUIPMENT SUPPORT STRUCTURE		
• Elect. Power	3.0	
• SPS	7.0	
• Tele. & Command	4.0	
• Comm.	4.0	
• Experiments	7.0	(25.0)
THERMAL CONTROL		
• Insulation	18.2	
• Coatings	2.5	
• Heat Pipes	31.0	
• Vanes (3)	6.9	
• RTG Radiators (3)	14.4	
• Isolators & Heaters	5.0	(78.0)
ELECTRICAL POWER SYSTEM		
• RTG (3)	240.0	
• Battery	37.0	
• Shunt Reg.	3.0	
• Fault Detector	6.0	
• Shunt Dissipators	3.4	
ELECTRICAL POWER SYSTEM (Continued)		
• Power Condit. Unit	17.6	
• Load Controller	3.0	
• Battery Cont.	5.0	
• Misc.	5.0	(320.0)
S/C PROPULSION AND ATTITUDE CONTROL SYSTEM		
• Tanks	25.0	
• Propellant	372.0	
• Sensors	26.0	
• Wheels (3)	58.0	
• Electronics	20.0	
• Computers	45.0	(546.0)
TELE. & COMMAND SYSTEM		
• S Band Rec	8.0	
• S Band Trans.	4.0	
• VHF Rec.	6.0	
• VHF Trans.	4.0	
• LDR Mux	6.0	
• MDR Mux	6.0	
• Recorders (2)	40.0	
• CMD Decoder	5.0	(79.0)
BASIC S/C WEIGHT		(1236.0)
MISSION EQUIPMENT		(764.0)
TOTAL SPACECRAFT		(2000.0)

This spacecraft has the form of an annulus four feet high with inner and outer diameters of 4 and 13 feet. As with the RTG bus, the central volume is occupied by the tanks and thrusters of a hydrazine monopropellant system used for orbit adjustment and attitude control. Vehicle weight including mission equipment can range up to 4000 pounds.

Internally, the spacecraft is divided into eight compartments, two occupied by the components of the Brayton power system, two by standard spacecraft subsystems, and four available for mission equipment. For the earth-viewing missions, one end of the cylinder is pointed toward the target and directional equipment is mounted on this face. The two faces of the cylinder provide mounting surfaces for the spacecraft equipment and dissipate heat originating within the equipment. As with the RTG bus, temperature sensitive components are mounted inside the spacecraft; large temperature-insensitive components outside.

The complete power plant, including the radiator, is assembled as a unit and its operation tested before mating to the rest of the spacecraft. The radiator consists of the outer cylindrical skin of the spacecraft which is slipped over the structural members during assembly. Relatively uniform radiator temperature is achieved by dividing the Brayton working fluid into two streams which encircle the radiator in opposite directions. Figure S-20 shows typical temperature profiles around the radiator circumference resulting from this counterflow arrangement. Heat is distributed over the entire radiator surface by axial heat pipes. The heat is radiated both outward to space and inward to the payload for thermal control.

The thermal control system differs from that of the RTG bus in that heat input from the radiator is not varied with different conditions. However, it still provides environmental conditions between -10°C and +40°C for all spacecraft equipment in all conditions. Thus the Brayton bus is adaptable to arbitrary earth-orbit mission conditions, even though the mission model listed only earth-facing applications in the power range served by this spacecraft. Figure S-21 shows the equipment panel temperature range for various power levels and mission conditions.

Some of the subsystems, such as the propulsion and attitude control, are substantially identical to those of the RTG bus except for size. The fixed-nozzle thrusters still require spinning the spacecraft to accomplish large velocity increments, and the configuration is arranged to give moments of inertia which provide stable spinning about the thrust axis. The telemetry system is identical to that of the RTG bus. The AC power output of the Brayton alternator is converted to 28 volt DC, regulated to ± 2%. The power system must also provide parasitic loads to keep the Brayton unit operating at a constant speed of rotation despite variations in equipment load.

Table S-9 is a weight breakdown for a 4000 pound Brayton bus with three fuel capsules powering the Brayton unit and containing 560 pounds of propellant. The mission equipment allowance is almost 1900 pounds.

The adaptability of the standard Brayton spacecraft to two specific missions was confirmed by detailed studies. Figure S-22 is a sketch of an Earth Observation Satellite (EOS) mission deployed on a Brayton bus. Figure S-23 shows an Applications Technology Satellite ATS-G mission deployed on the bus; Figure S-24 shows the same equipment stowed on the bus.

Figure S-25 shows the Brayton bus mounted to the orbiter payload compartment via a pallet, and Figure S-26 shows both RTG and Brayton buses on the same pallet within the orbiter. Figure S-27 shows how the pallet can be mounted to a tug vehicle for delivery to synchronous or other orbits.

3.5 Reactor Thermoelectric System Applications.

Figure S-28 is a layout of a side-looking radar mission system mounted to a reactor thermoelectric system. The 25-foot long by 2.6 foot high slotted-waveguide-array antenna scans a 45-nautical-mile swath on the earth for mapping purposes. The orbital altitude is 300 nautical miles. With 5 kW(e) of power available, a 40-foot range resolution becomes possible. Similar performance can be achieved from higher altitudes by increasing the antenna height.

An AEC study confirmed the app[illegible] bility of the same reactor system to a high-powered synchronous orbit communications mission. (Ref. S-3).

TABLE S-9

BRAYTON BUS - WEIGHT ESTIMATE

PRIMARY STRUCTURE		
• Shell	113.0	
• Ribs	55.5	
• Rings	83.5	
• Stiffeners	7.3	
• Brayton Spt. Str.	18.0	
• Bus Spt. Str.	90.0	
• Contig (10%)	37.0	(404.3)
EQUIPMENT SUPPORT STRUCTURE		
• Elect. Power	5.0	
• SPS	10.0	
• Tele. & Command	5.0	
• Comm.	5.0	
• Experiments	10.0	(35.0)
THERMAL CONTROL		
• Heat Pipes	54.0	
• Insulation	35.0	
• Coatings	17.0	
• Paint	5.7	
• Heaters	5.0	
• Misc. Isolators	5.0	(121.7)
ELECTRICAL PWR. SYS. (BRAYTON UNIT)		
• Gas Radiator	92.0	
• Heat Sources (3)	305.0	
• Recuperator	68.0	
• Turbine	25.0	
• Plumbing	12.0	
• Electronics	15.0	
• Batt. & Inverter	59.0	
• PLR	20.0	
• Insulation	109.2	(705.2)

PROPULSION AND ATTITUDE CONTROL SYSTEM		
• Tanks	32.0	
• Propellant	580.0	
• Sensors	17.0	
• Wheels (3)	84.0	
• Computers (2)	45.0	
• Electronics	20.0	(758.0)
TELE. & COMMAND SYSTEM		
• S Band Rec.	8.0	
• S Band Trans.	4.0	
• VHF Rec.	6.0	
• VHF Trans.	4.0	
• LDR Mux	6.0	
• MDR Mux	6.0	
• CMD Decoder	5.0	
• Recorders (2)	40.0	(79.0)
BASIC S/C WEIGHT		(2103.2)
MISSION EQUIPMENT		(1896.8)
TOTAL SPACECRAFT		(4000.0)

• Bus Spt. Str.	90.0	
• Contig (10%)	37.0	(404.3)
EQUIPMENT SUPPORT STRUCTURE		
• Elect. Power	5.0	
• SPS	10.0	
• Tele. & Command	5.0	
• Comm.	5.0	
• Experiments	10.0	(35.0)
THERMAL CONTROL		
• Heat Pipes	54.0	
• Insulation	35.0	
• Coatings	17.0	
• Paint	5.7	
• Heaters	5.0	
• Misc. Isolators	5.0	(121.7)
ELECTRICAL PWR. SYS. (BRAYTON UNIT)		
• Gas Radiator	92.0	
• Heat Sources (3)	305.0	
• Recuperator	68.0	
• Turbine	25.0	
• Plumbing	12.0	
• Electronics	15.0	
• Batt. & Inverter	59.0	
• PLR	20.0	
• Insulation	109.2	(705.2)

• Electronics	20.0	(758.0)
TELE. & COMMAND SYSTEM		
• S Band Rec.	8.0	
• S Band Trans.	4.0	
• VHF Rec.	6.0	
• VHF Trans.	4.0	
• LDR Mux	6.0	
• MDR Mux	6.0	
• CMD Decoder	5.0	
• Recorders (2)	40.0	(79.0)
BASIC S/C WEIGHT		(2103.2)
MISSION EQUIPMENT		(1896.8)
TOTAL SPACECRAFT		(4000.0)

4 COST STUDIES

Cost studies were carried out to compare standardized nuclear-powered spacecraft with mission-specialized (or "dedicated") solar-powered spacecraft. Attention was limited to ten missions from the NASA 1972 mission model which are now planned for the post 1979 period: five below 500 watts and five above 500 watts, i.e. in the RTG and Brayton system power ranges.

Launch costs, operational costs, and specialized mission equipment costs were not considered in this phase of the study. No detailed analysis was made for the cost increases for handling nuclear fuel, nor for reduction in launch and operational costs due to standardization. These two factors were assumed to offset each other. Mission equipment costs should be relatively unaffected by the nature of the spacecraft. In view of the relative imprecision of cost models, it was assumed that these particular costs are the same for both types of spacecraft, and attention was limited to the actual spacecraft costs and to the costs of the standard equipment provided for all missions.

The cost model was drawn from the section of the USAF Space Planners Guide dealing with space vehicle system costs. It divides the costs into non-recurring costs, involving development, test, and evaluation (DTE), facilities, and aerospace ground equipment (AGE), and recurring costs of production and operations. The operations costs are not included.

Within each category, costs are broken down by subsystems, and estimated from a set of curves which usually plot cost against weight. In most cases several different curves are furnished to cover different types of each particular subsystem. While the subsystem cost estimates may not be accurate, the total system costs do provide a reasonable reflection of existing system costs. The model is thus useful for quick estimates of total system costs during early stages of conceptual design.

Since the model was prepared in 1965, costs were first increased by 30 percent to reflect inflation. The non-recurring DTE costs were then decreased by 25 percent to reflect the savings due to the NASA subsystem standardization program. The estimated weight of the dedicated spacecraft was assumed to be less than that of the

standardized designs, since the former could presumably be more efficient in weight utilization by virtue of addressing only one mission rather than providing capability for a variety of missions. In the USAF cost models, this weight difference imposed a penalty on the standardized spacecraft, since all the spacecraft subsystem cost curves showed increasing cost with increasing weight.

Recurring costs were also estimated as a function of weight (or power, for the power subsystem only), with an initial unit cost modified by a set of "learning curves" to reflect the effects of the size of production runs. In general, the recurring costs for the standardized spacecraft are higher than those for the dedicated spacecraft, while the non-recurring costs are lower, since most of the DTE spacecraft costs were eliminated by the use of a standard design.

The comparison for four lower-powered missions serviced by the standard RTG spacecraft is shown in Table S-10. The Tracking and Data Relay Satellite mission (TDRS), which required less than 500 watts, was found to be accomplished at lower cost by use of the Brayton S/C. The table shows the pattern of higher recurring costs for the standardized spacecraft, and lower DTE costs. The final totals of $74 million for the RTG spacecraft and $218 million for the dedicated spacecraft show a savings of 66 percent from standardization. The total savings of $144 million on these four missions more than amortize the estimated $81 million cost of developing the standard RTG spacecraft.

Cost comparisons for six medium-power missions appear in Table S-11, which estimates total costs for a Brayton-powered standard spacecraft and for a solar-powered dedicated spacecraft. This table shows overall costs of $341 million for the standardized spacecraft and $636 million for the dedicated spacecraft, or a net saving of 46 percent. The net savings of $295 million completely amortize, over only a few missions, the $105 million estimated cost of developing the Brayton spacecraft. The somewhat higher recurring costs of the standard spacecraft are more than offset by the savings in DTE. The non-recurring costs shown for the standard spacecraft consist principally of the cost of integrating each mission with the spacecraft.

TABLE S-10

COST COMPARISON FOR FOUR LOW-POWER MISSIONS*
STANDARDIZED RTG S/C VERSUS SPECIALIZED SOLAR-POWERED S/C
In Millions of Dollars**

MISSION	Upper Atmos. Eplor.	Synch. Meteor. Sat.	TIROS	Geopause	TOTAL
FLIGHTS	6	2	1	2	11
STANDARD S/C (Nuclear)					
Non-Recurring	0.9	1.5	4.4	5.3	12.1
Recurring	32.6	10.9	7.8	10.9	62.2
TOTAL	33.5	12.4	12.2	16.2	74.3
SPECIALIZED S/C (Solar)					
Non-Recurring	53.4	37.1	50.5	48.5	189.5
Recurring	15.5	3.7	3.7	5.7	28.6
TOTAL	68.9	40.8	54.2	54.2	218.1
RTG S/C SAVINGS	35.4	28.4	42.0	38.0	143.8
	51%	70%	77%	70%	66%
COST OF DEVELOPING STANDARD S/C					81.0
NET SAVINGS, After Full Amortization Over These Four Missions of the Standard S/C Development Cost					62.8 29%

* Costs Include: Standard S/C Subsystems (Structure, Thermal, Power, Attitude Control, Telemetry, Command), AGE, and DTE

Costs Exclude: Specialized Mission Equipment, Launch, and Operation

** Based on USAF Space Planners Guide Cost Formula, Plus 30% for Inflation, Component and Subsystem Standardization Assumed to Reduce DTE Costs by 25%

TABLE S-11

COST COMPARISON FOR SIX MEDIUM-POWER MISSIONS*
STANDARDIZED BRAYTON S/C VERSUS SPECIALIZED SOLAR-POWERED S/C
In Millions of Dollars**

MISSION	EOS	SEOS	ERS	Disaster Warning Satellite	System Test Sat.	TDRS	TOTAL
FLIGHTS	7	5	8	2	8	6	36
STANDARD S/C (Nuclear)							
Non-Recurring	6.6	20.2	20.0	6.5	11.1	2.2	66.6
Recurring	56.8	50.1	64.9	16.2	49.7	37.2	274.9
TOTAL	63.4	70.3	84.9	22.7	60.8	39.4	341.5
SPECIALIZED S/C (Solar)							
Non-Recurring	79.5	78.7	79.4	74.3	103.8	38.4	454.1
Recurring	37.7	25.5	43.1	11.0	54.4	10.7	182.4
TOTAL	117.2	104.2	122.5	85.3	158.2	49.1	636.5
BRAYTON S/C SAVINGS	53.8	33.9	37.6	62.6	97.4	9.7	295.0
	46%	33%	31%	73%	61%	20%	46%
COST OF DEVELOPING STANDARD S/C							105.0
NET SAVINGS, After Full Amortization Over These Six Missions of the Standard S/C Development Cost							189.7 30%

*Costs Include: Standard S/C Subsystems (Structure, Thermal, Power, Attitude Control, Telemetry, Command), AGE, and DTE

Costs Exclude: Specialized Mission Equipment, Launch, and Operation

**Based on USAF Space Planners Guide Cost Formula, Plus 30% for Inflation, Component and Subsystem Standardization Assumed to Reduce DTE Costs by 25%

More detailed cost studies using actual cost estimates instead of correlative models must await more detailed knowledge of the missions to be flown than is currently available, and a clearer definition of the costs of shuttle utilization and how these costs are to be apportioned among users.

The net result of the cost studies is to show that standardized vehicles using nuclear power offer distinct savings over specialized solar-powered vehicles designed for each mission. In general, the fewer the number of flights in a given mission, the greater the percentage of savings provided by use of standardized spacecraft. The major saving occurs in the elimination of much of the expensive development, test, and evaluation which accompanies the use of a specialized vehicle.

REFERENCES

Ref. S-1 Aerospace Report ATR-72 (7231)-1, Vol. II (Aug. 1971) Integrated Operations/Payloads/Fleet Analysis Final Report.

Ref. S-2 Aerospace Report ATR-72(7312)-1, Vol. II (Aug. 1972) "NASA Payload Data Book, Payload Analysis for Space Shuttle"

Ref. S-3 NASA CR 120873, "Applications Technology Satellite Advanced Missions Study," Final Report, Contract NAS 3-14360, Fairchild Industries, Appendix A.

LIST OF ACRONYMS AND ABBREVIATIONS

AGE	Aerospace Ground Equipment
ATS	Applications Technology Satellite
AVCS	Attitude and Velocity Control System
BOL	Beginning of Life
BRU	Brayton Rotating Unit
CG	Center of Gravity
CMD	Command
DOD	Department of Defense
DTE	Development, Test, and Evaluation
EM	Electromagnetic
EOM	End of Mission
EOS	Earth Observation Satellite
HSHX	Heat Source Heat Exchanger
LDRMux	Low Data Rate Multiplexer
LES	Lincoln Laboratory Experimental Satellite
MBS	Million Bits per Second
MDRMux	Medium Data Rate Multiplexer
MHW	Multi Hundred Watt
N. Mi.	Nautical Miles
NVT	Neutron Velocity Time
RIB	Radioisotope Brayton system
RTG	Radioisotope Thermoelectric Generator
RTS	Reactor Thermoelectric System
S/C	Spacecraft
SEOS	Synchronous Earth Observation Satellite
TDRS(s)	Tracking and Data Relay Satellite (system)
TE	Thermoelectric
TIROS	Television Infrared Observation Satellite
TTC	Tracking, Telemetry and Command
VHF	Very High Frequency

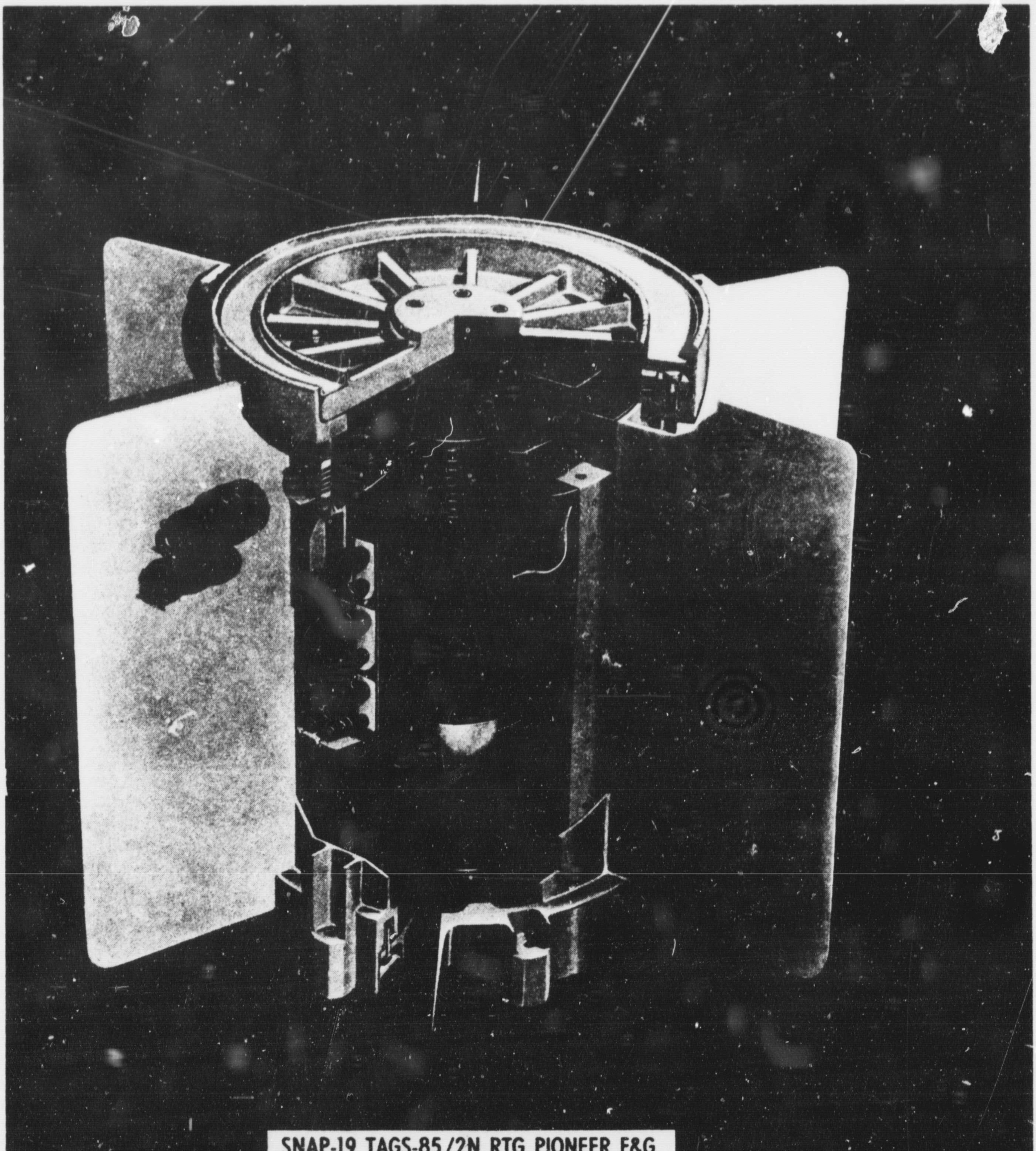


FIGURE S - 1

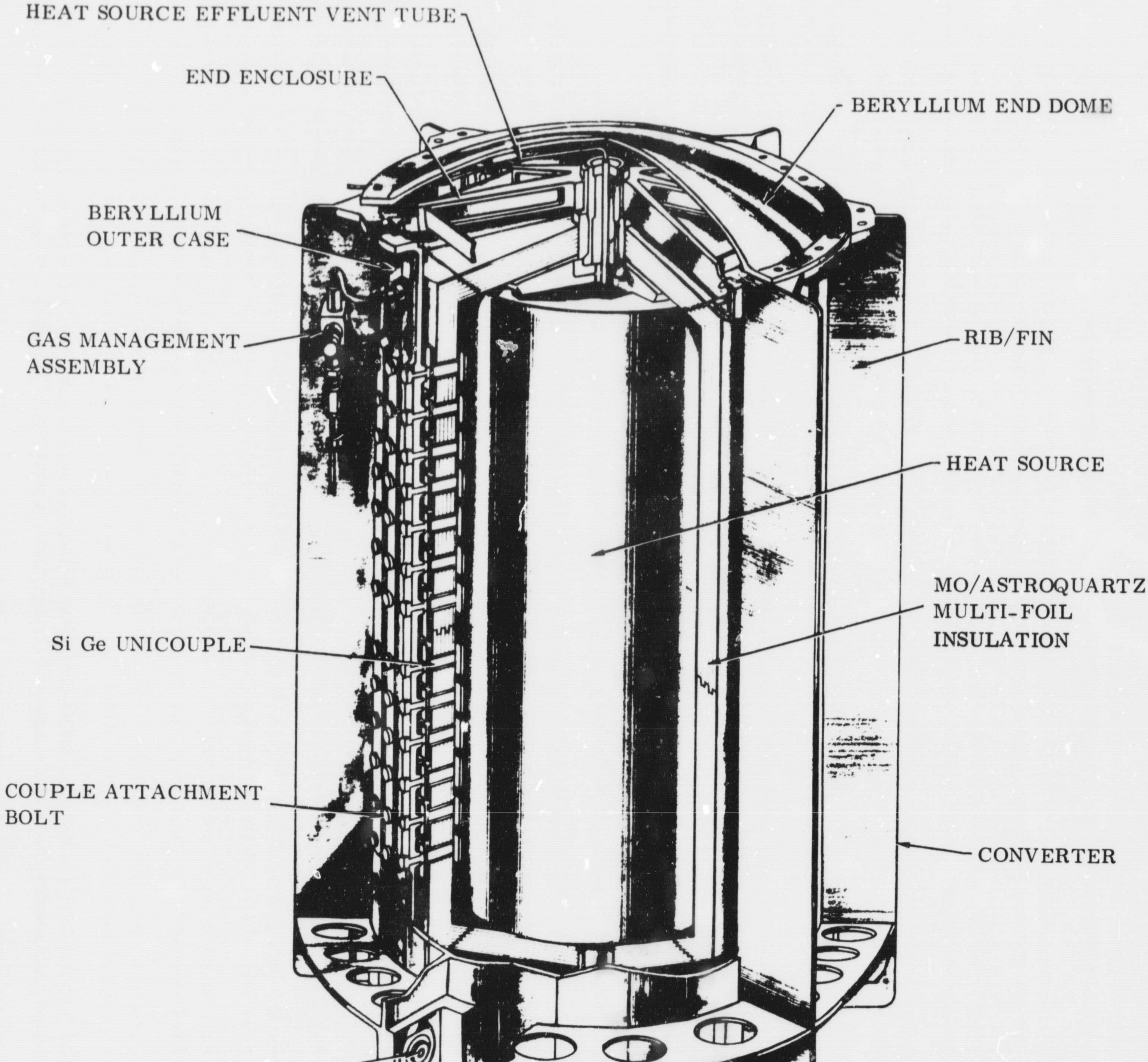


FIGURE S - 2 MHW RADIOISOTOPE THERMOELECTRIC GENERATOR

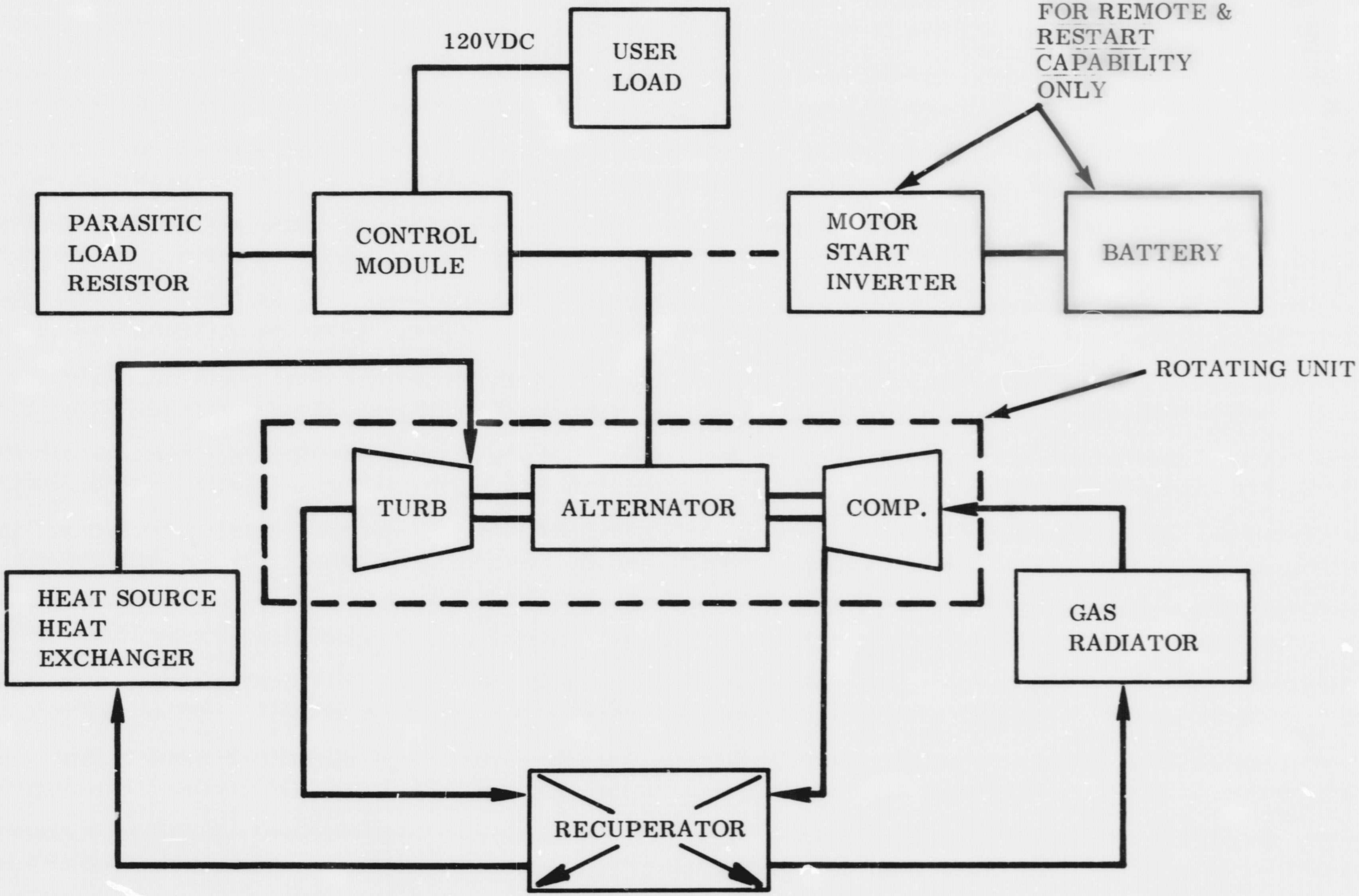


FIGURE S - 3 MINI-BRAYTON SCHEMATIC

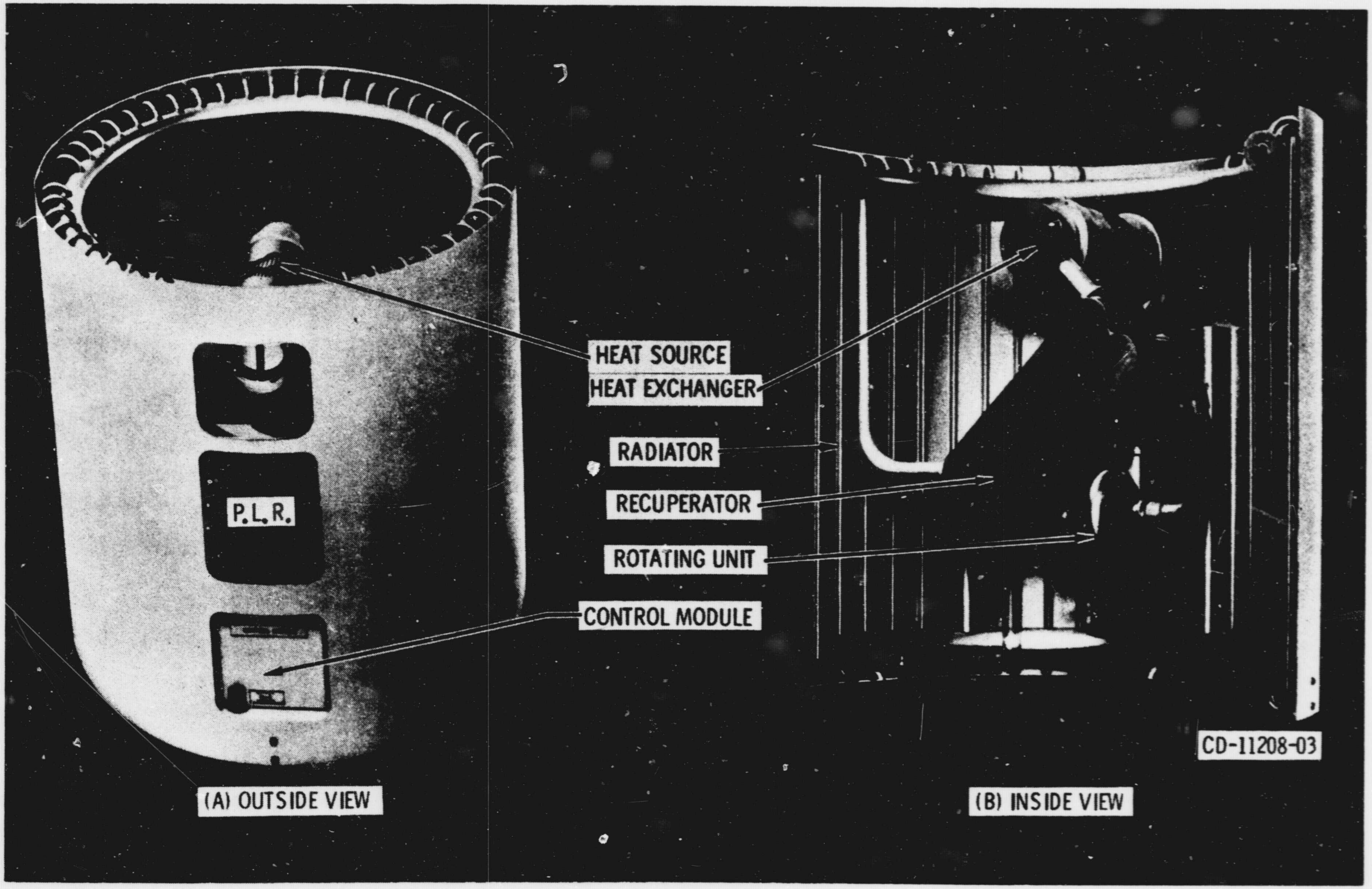


FIGURE S - 4 MINI-BRAYTON 550 WATT POWER SOURCE

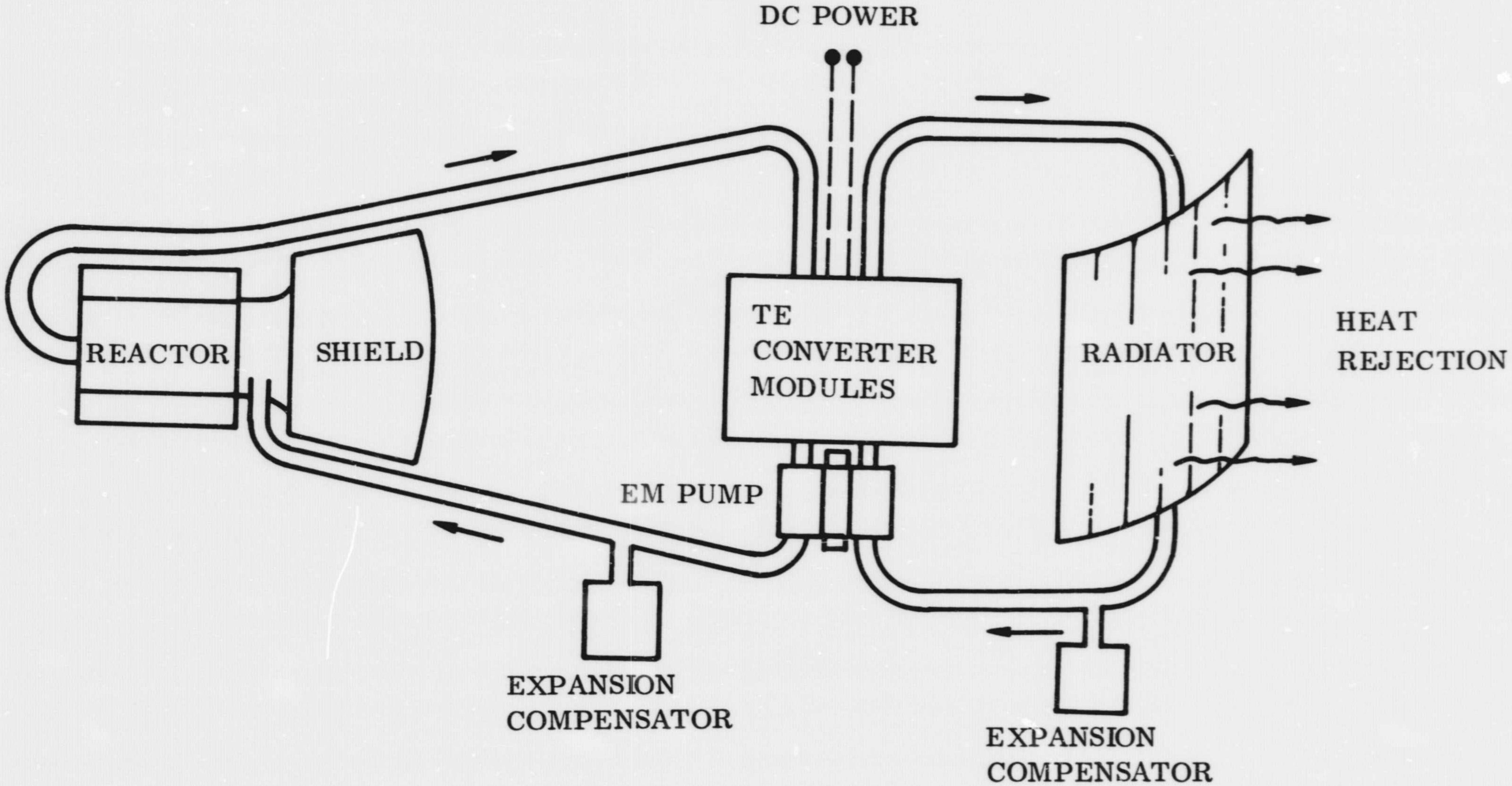


FIGURE S - 5 SCHEMATIC OF REACTOR - THERMOELECTRIC POWER SYSTEM

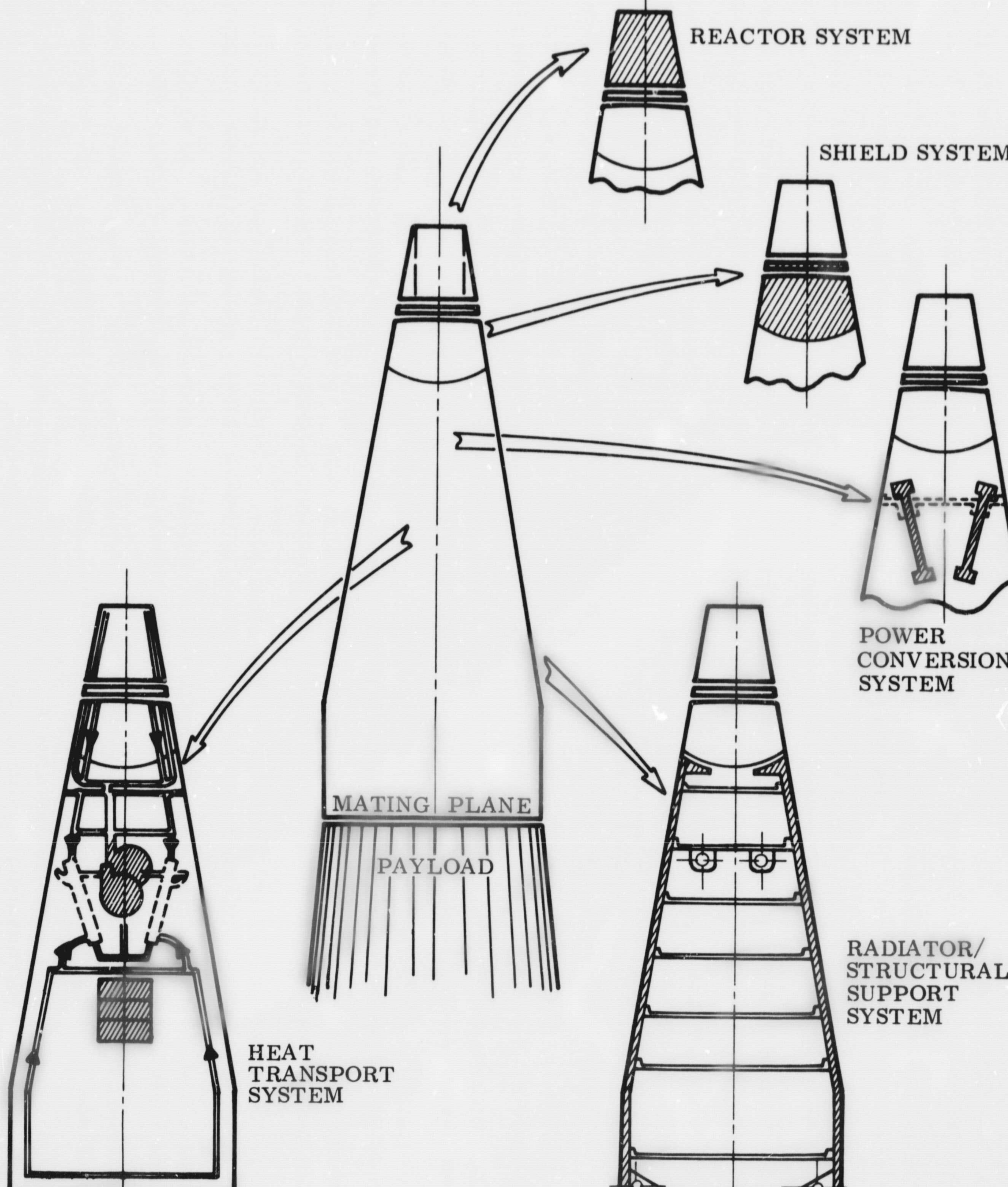
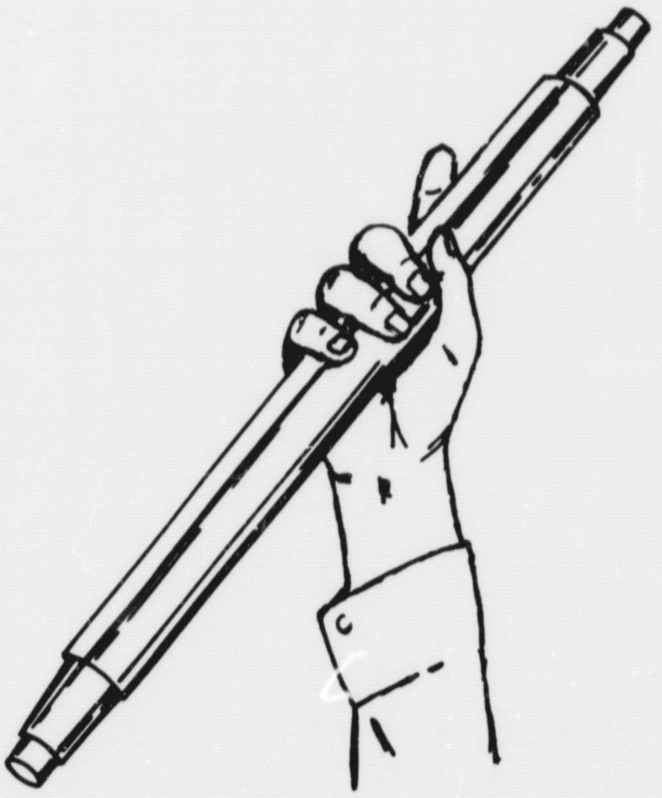


FIGURE S-6. ZrH REACTOR THERMOELECTRIC SPACE POWER SYSTEM FUNCTIONAL SCHEMATIC & FLOW DIAGRAM



● INNER DIAMETER (IN.)	0.75
● OUTER DIAMETER (IN.)	1.52
● ACTIVE LENGTH (IN.)	15.07

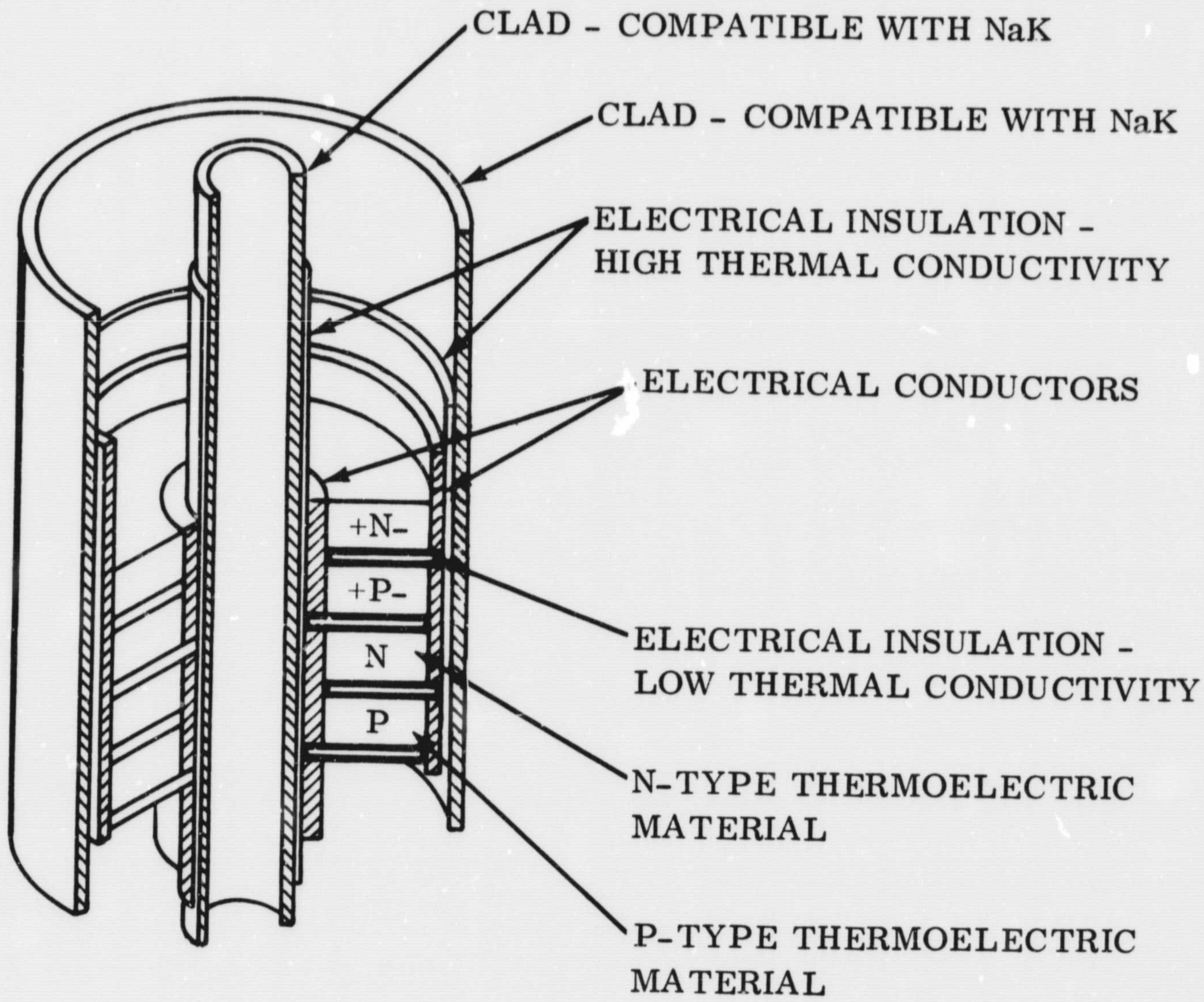


FIGURE S - 7 PbTe THERMOELECTRIC MODULE FOR REACTOR SYSTEM

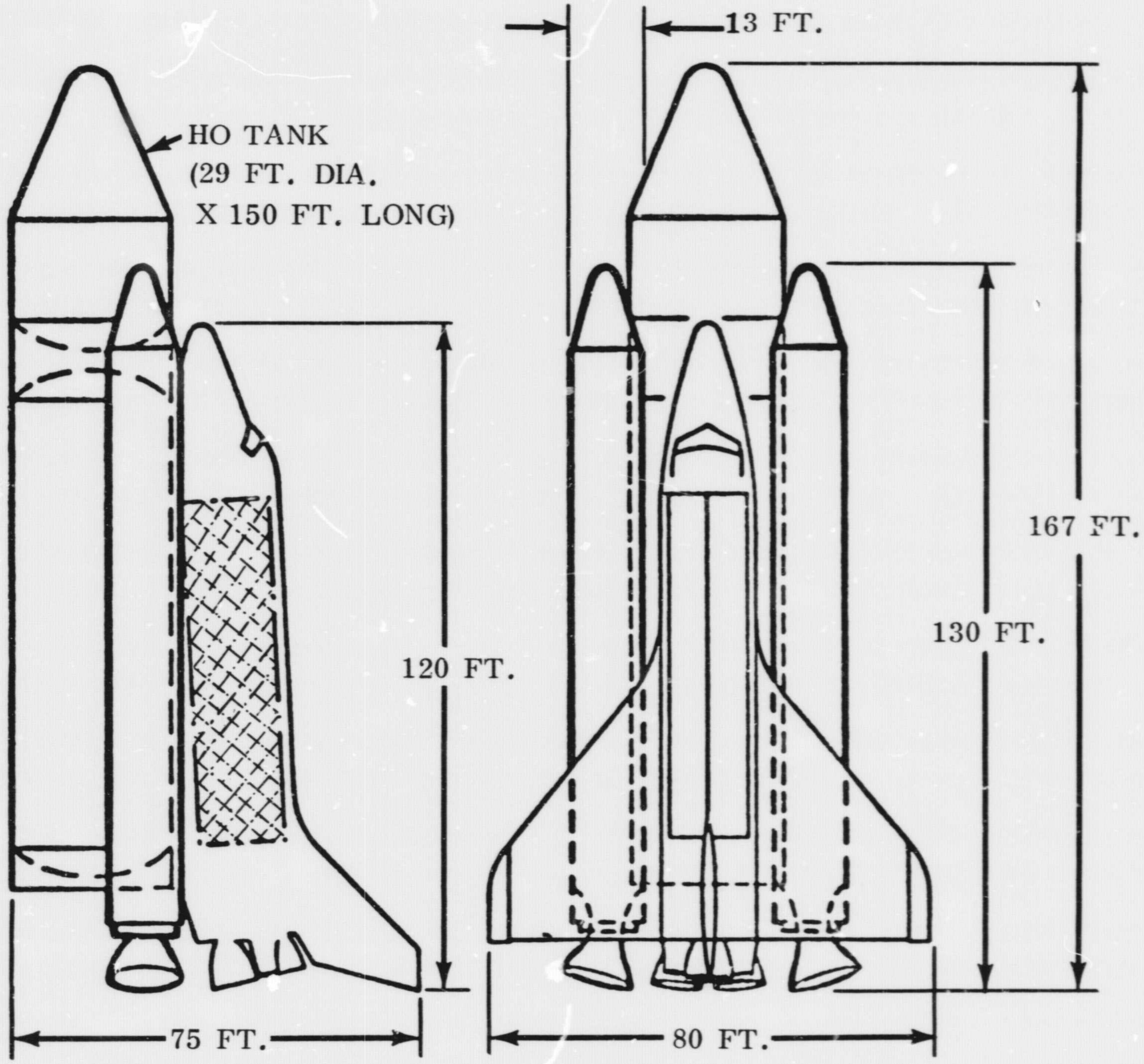


FIGURE S - 8 SPACE SHUTTLE SYSTEM

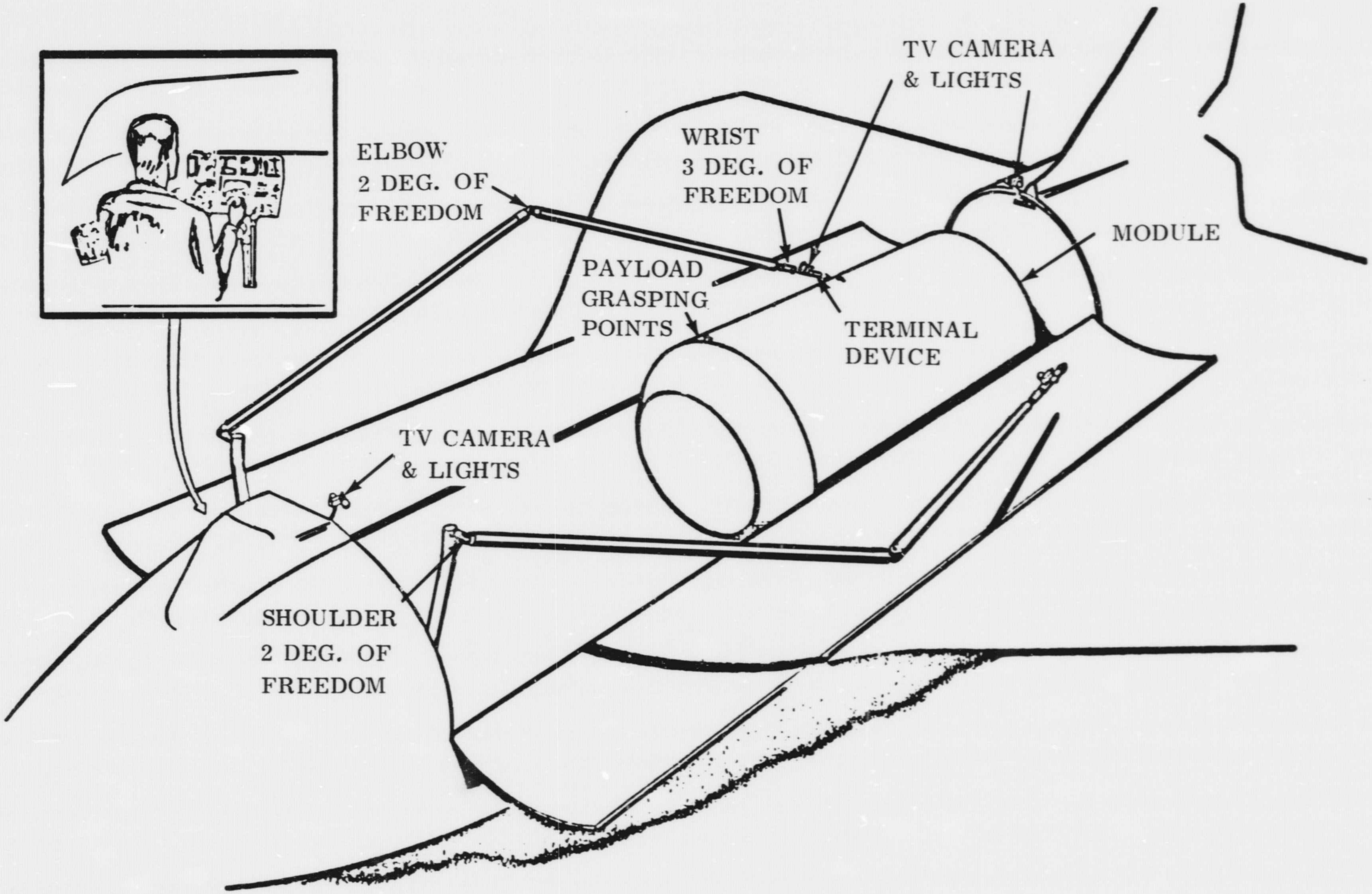


FIGURE S - 9 REMOTE MANIPULATOR SYSTEM

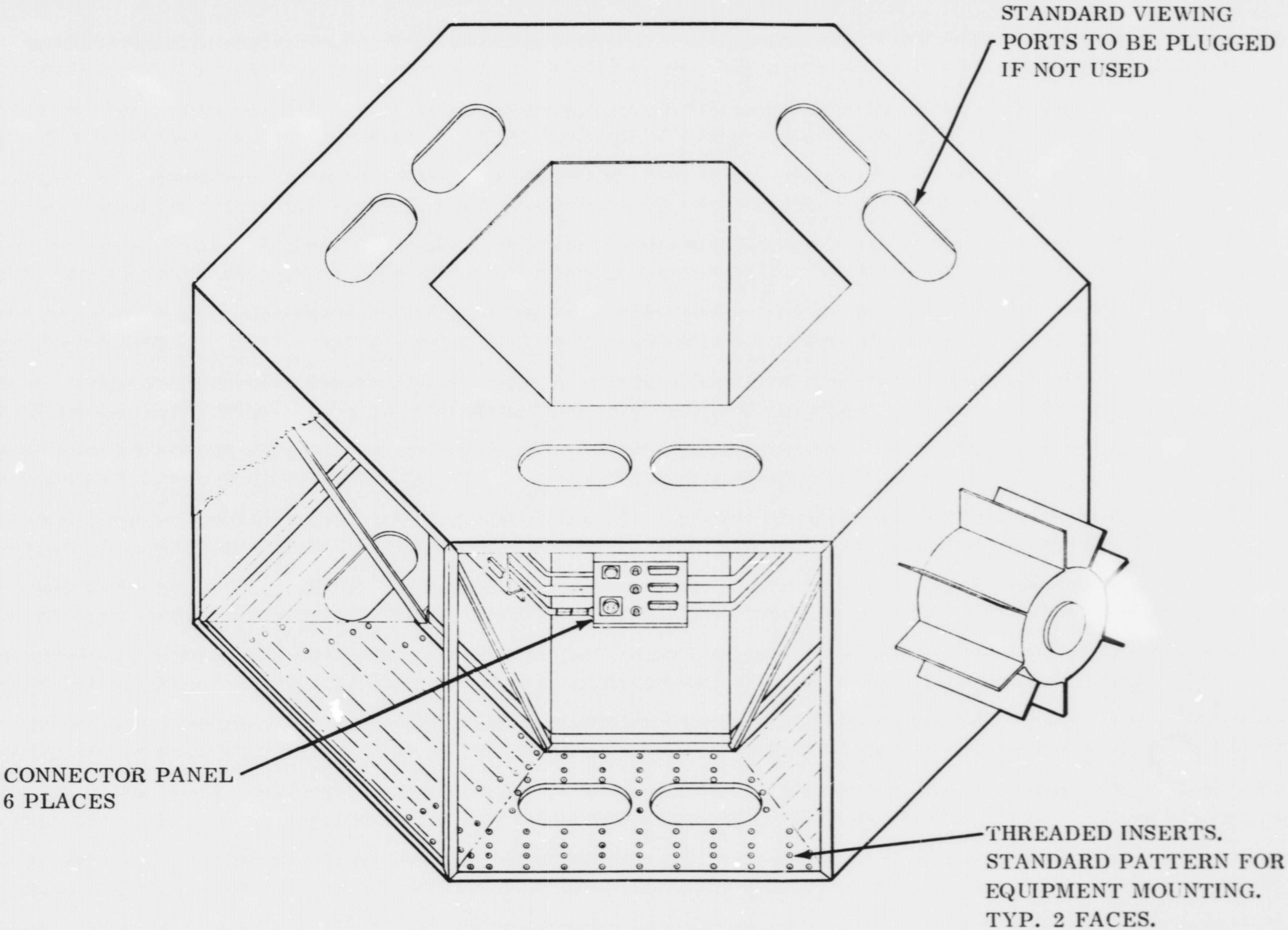
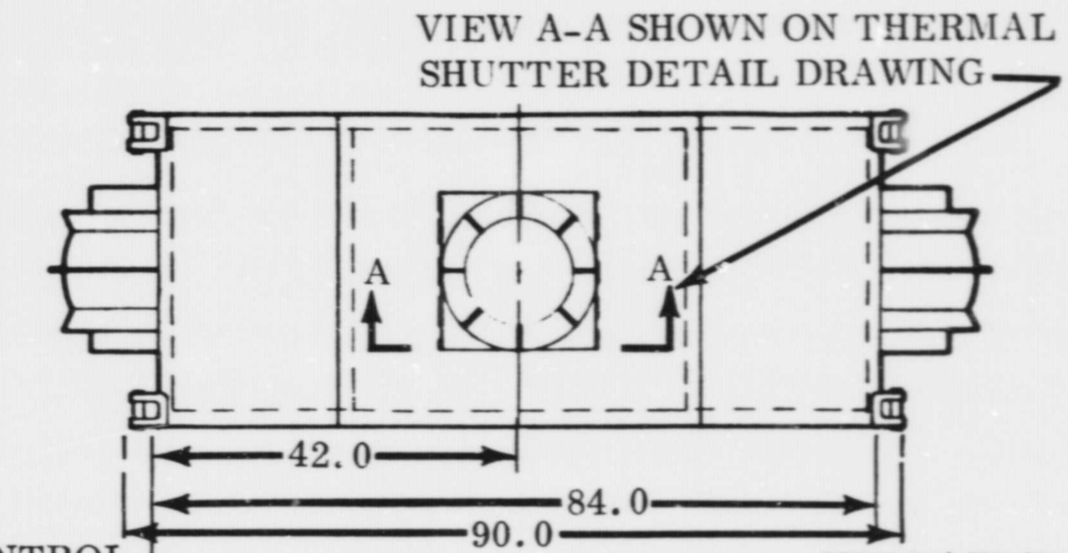


FIGURE S - 10 RTG S/C CUTAWAY SHOWING PAYLOAD COMPARTMENT

MISSION POWER REQUIREMENT	NUMBER RTG REQUIRED	LOCATION OF RTG
150 WATTS	1	LOCATION "A"*
300 WATTS	2	LOCATION "A" & "C"
450 WATTS	3	LOCATION "A," "B," & "D"

* FOR 150 WATT MISSION A BALLAST IS REQ'D AT LOCATION "C" FOR DYNAMIC BALANCE.



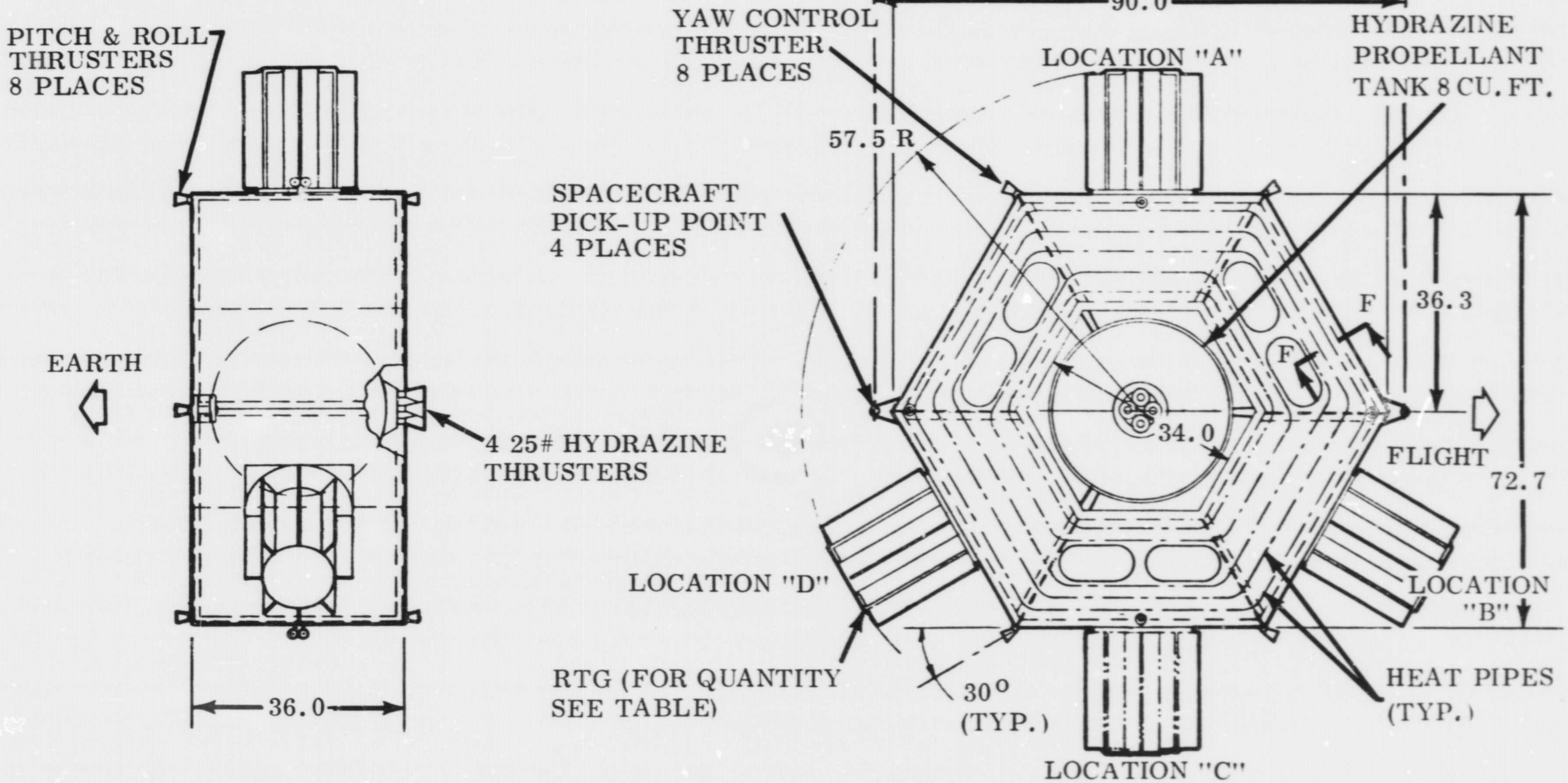


FIGURE S - 11 BASIC RTG S/C

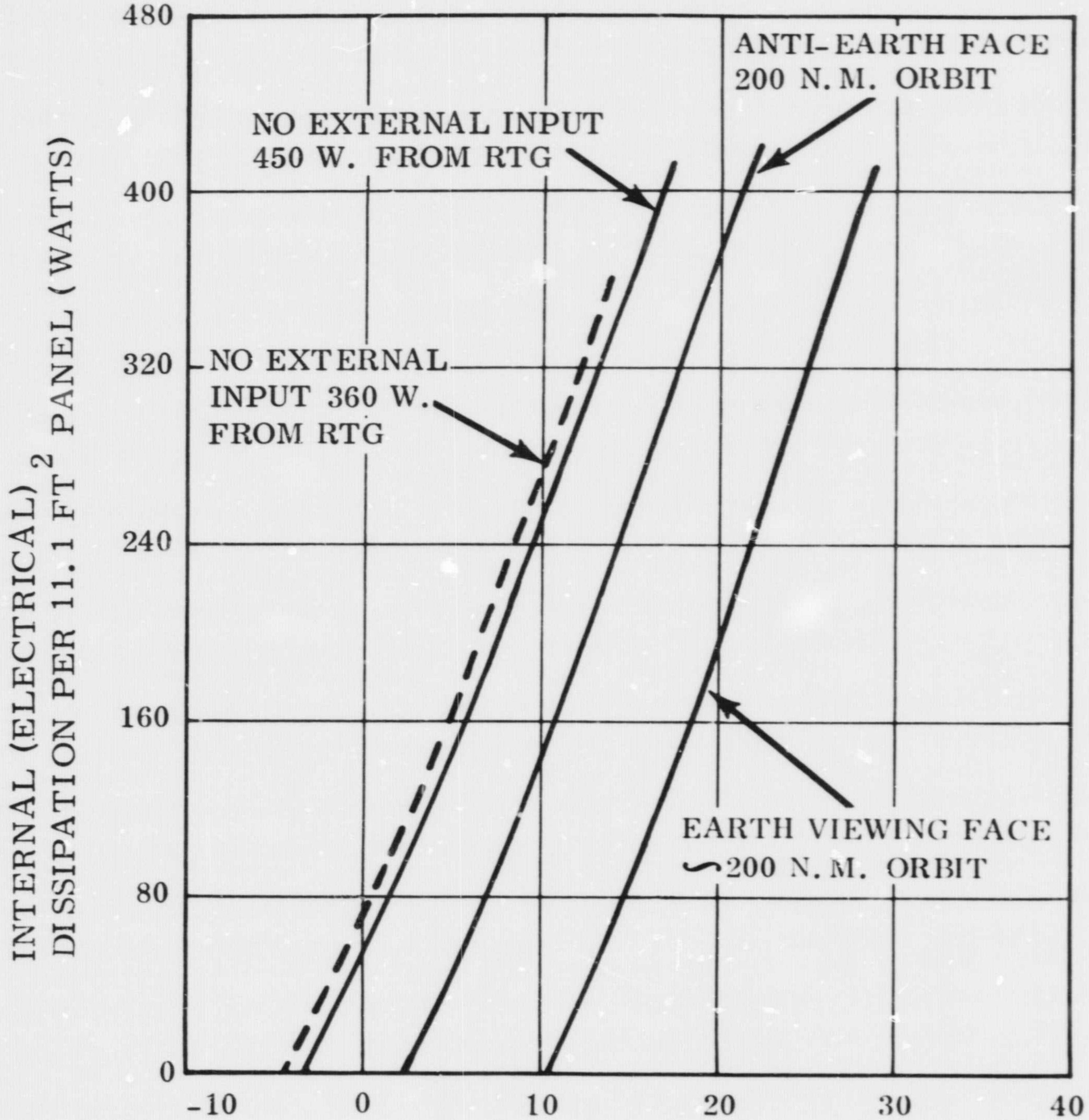


FIGURE S - 12 EQUIPMENT PANEL MEAN TEMPERATURES, RTG S/C

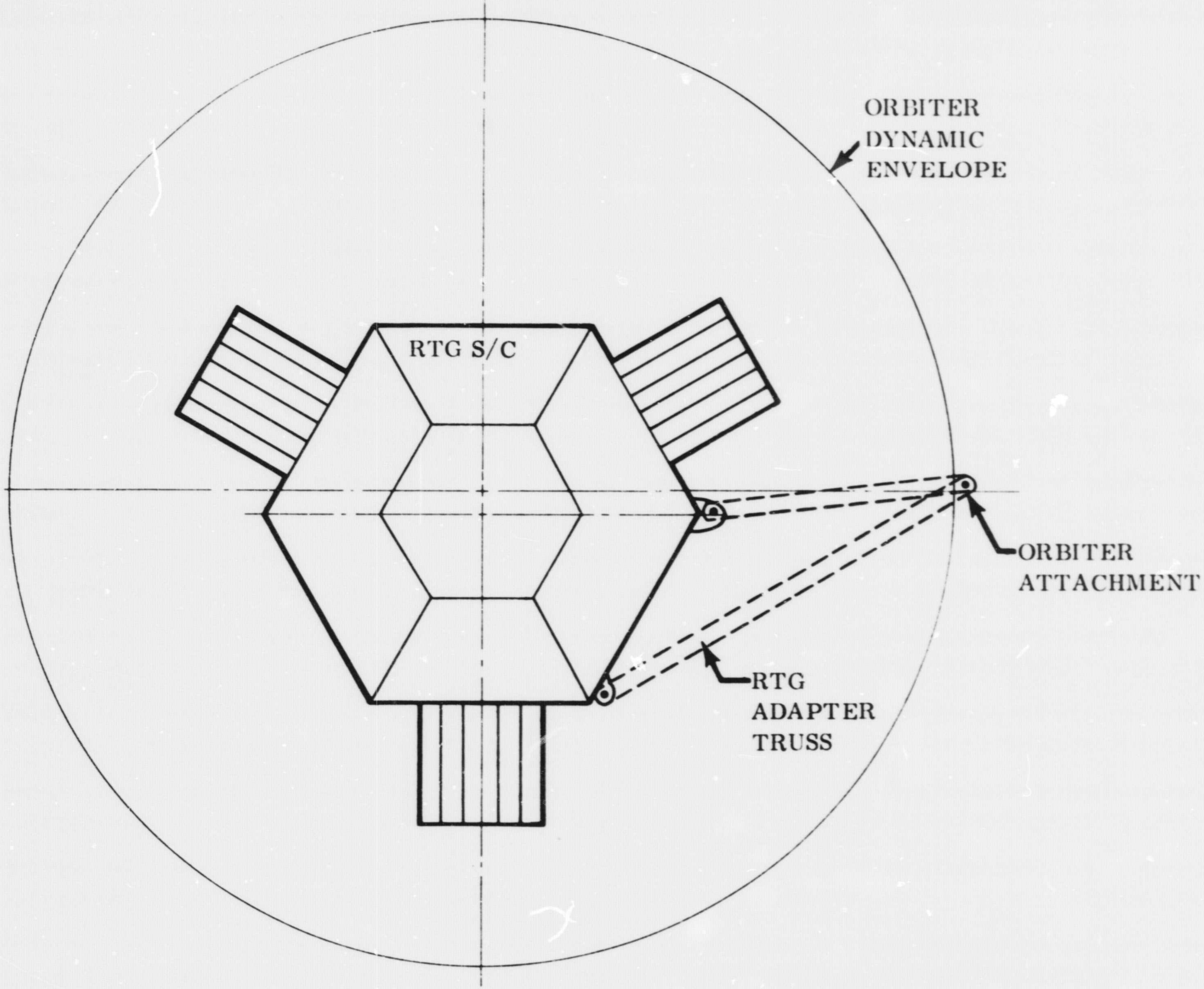


FIGURE S - 13 RTG S/C INSTALLATION TO ORBITER

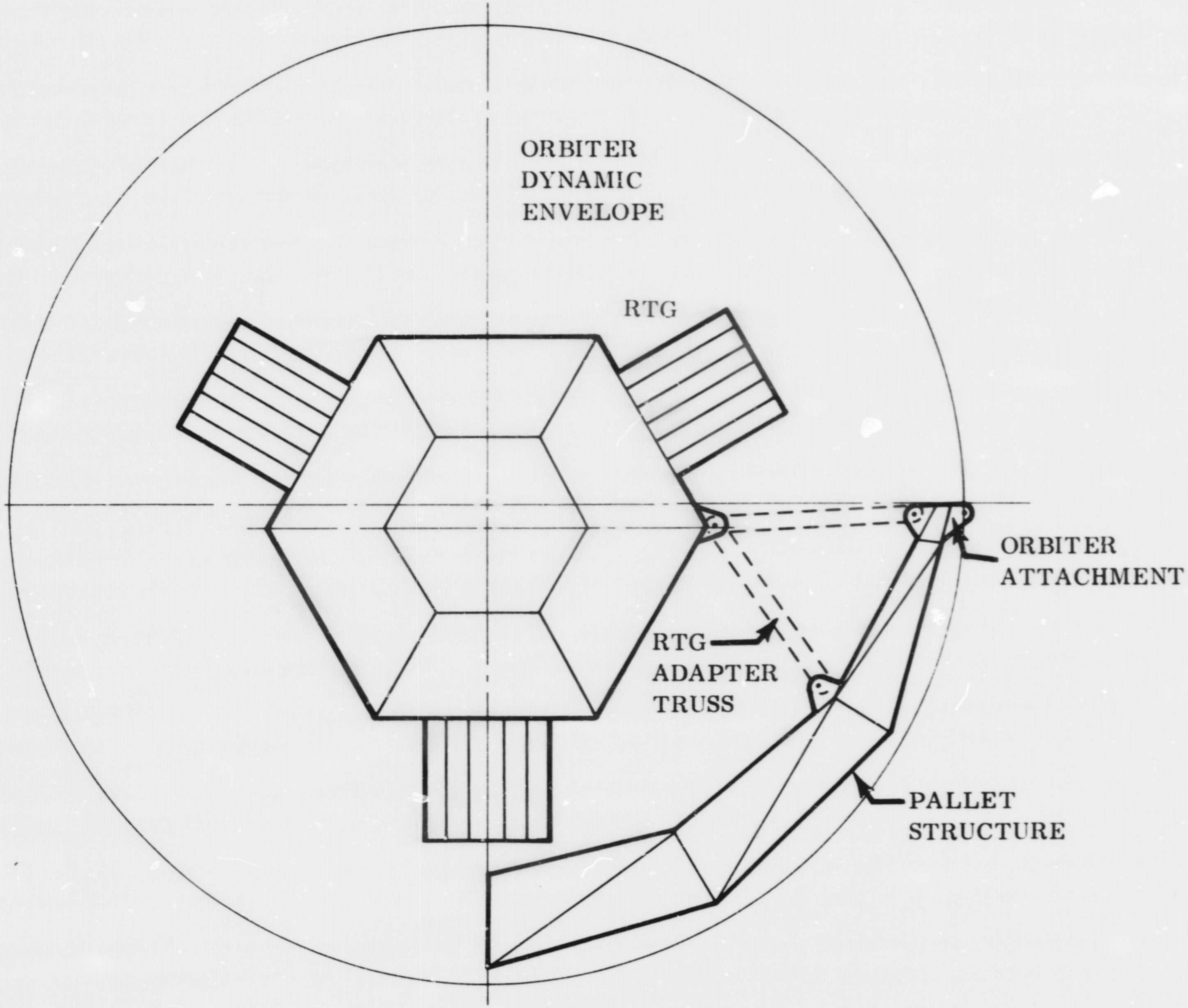


FIGURE S - 14 RTG S/C INSTALLATION ON PALLET

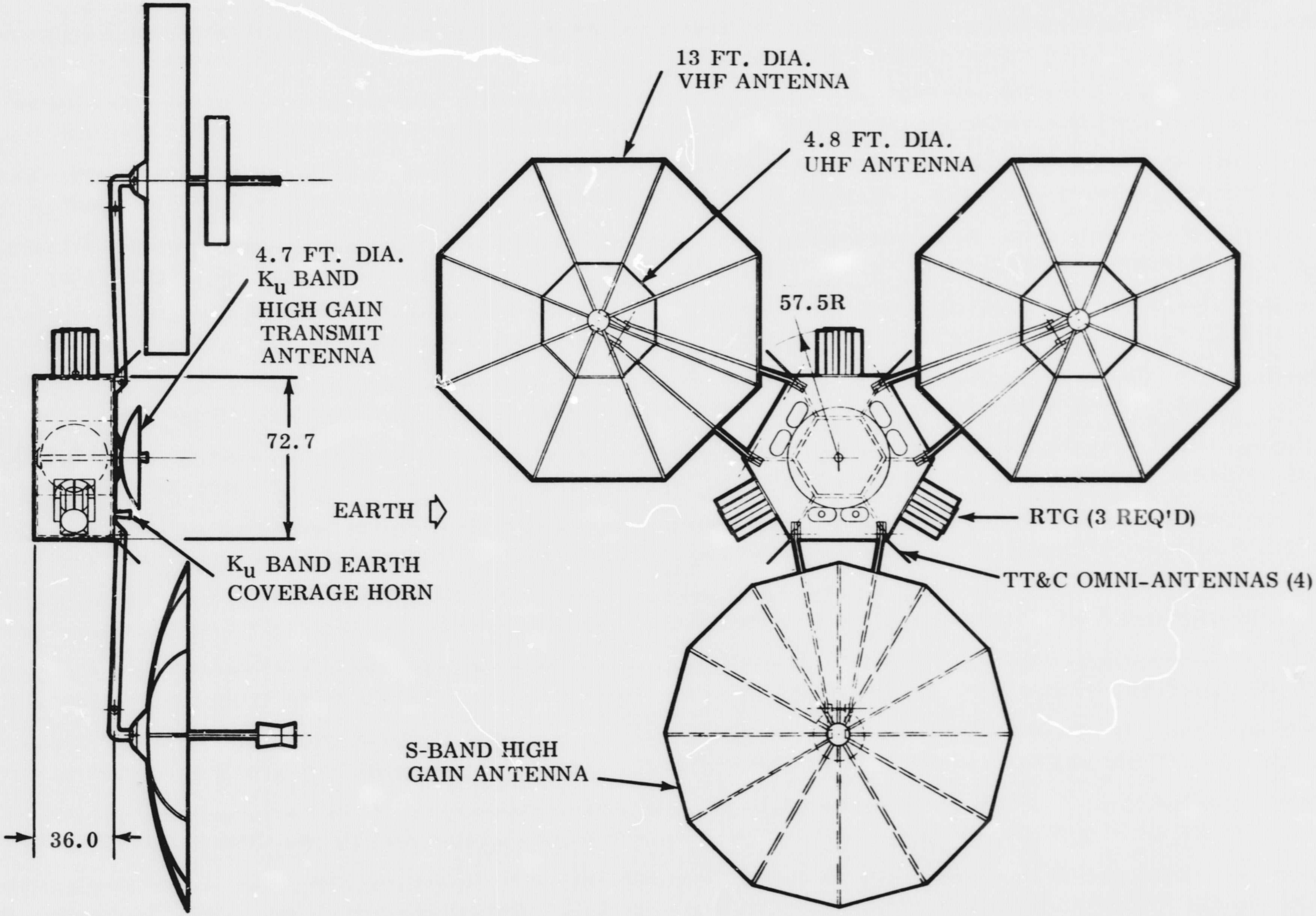


FIGURE S - 15 TRACKING & DATA RELAY SATELLITE SYSTEM (TDRSS) DEPLOYED ON RTG S/C

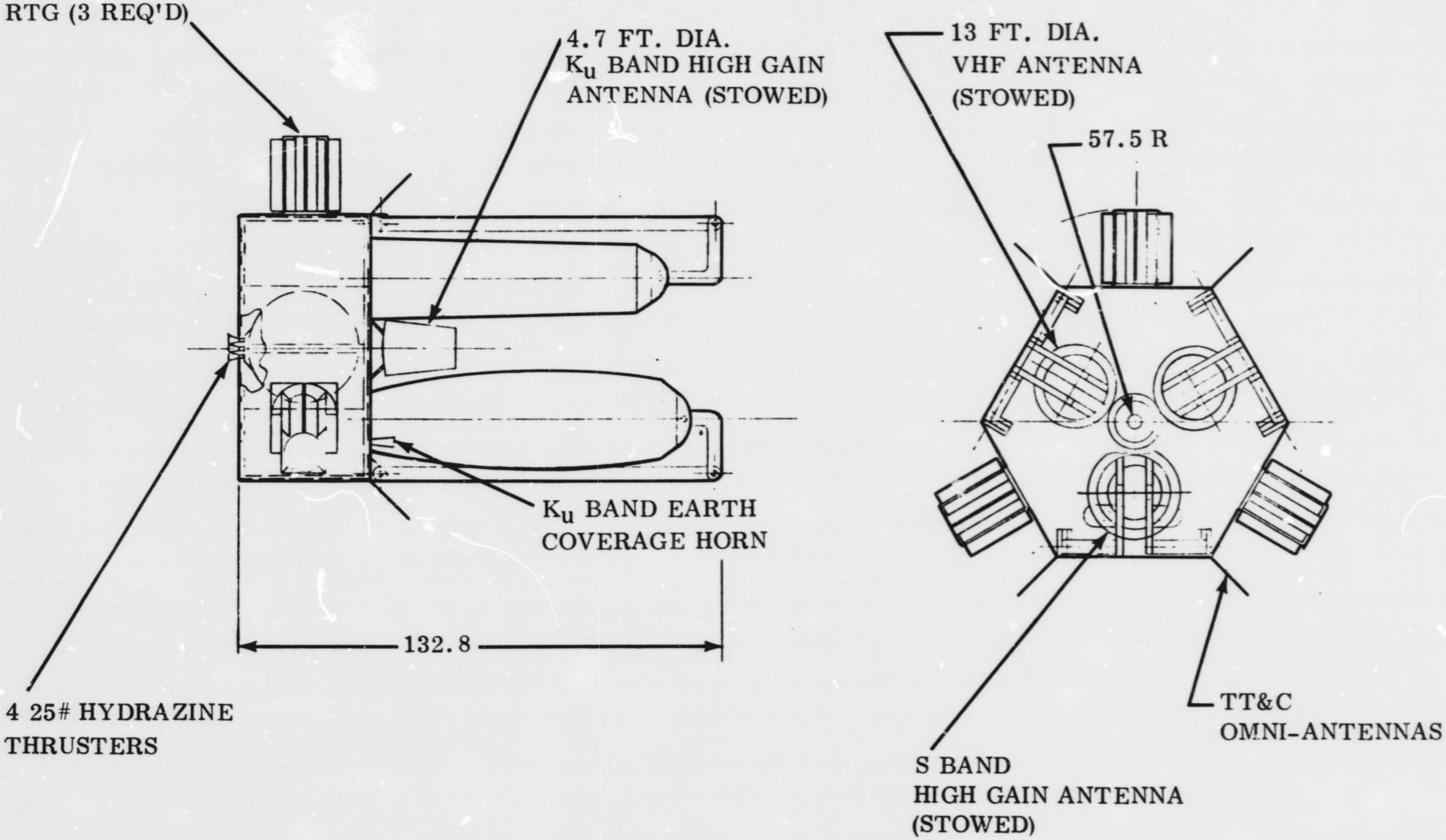


FIGURE S - 16 TDRSS STOWED ON RTG S/C

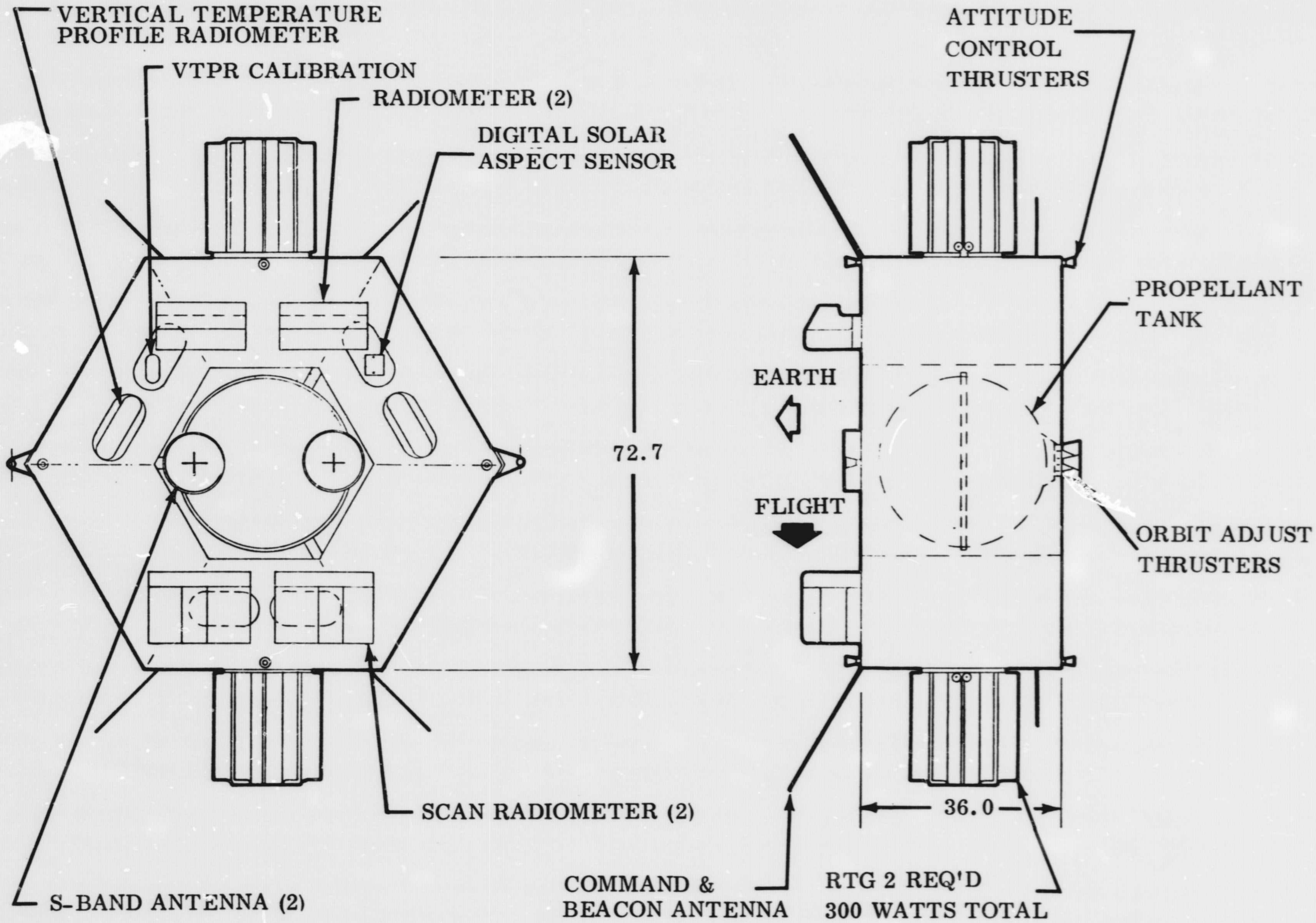


FIGURE S-17. TIROS FOLLOW-ON METEOROLOGICAL SATELLITE ON RTG S/C

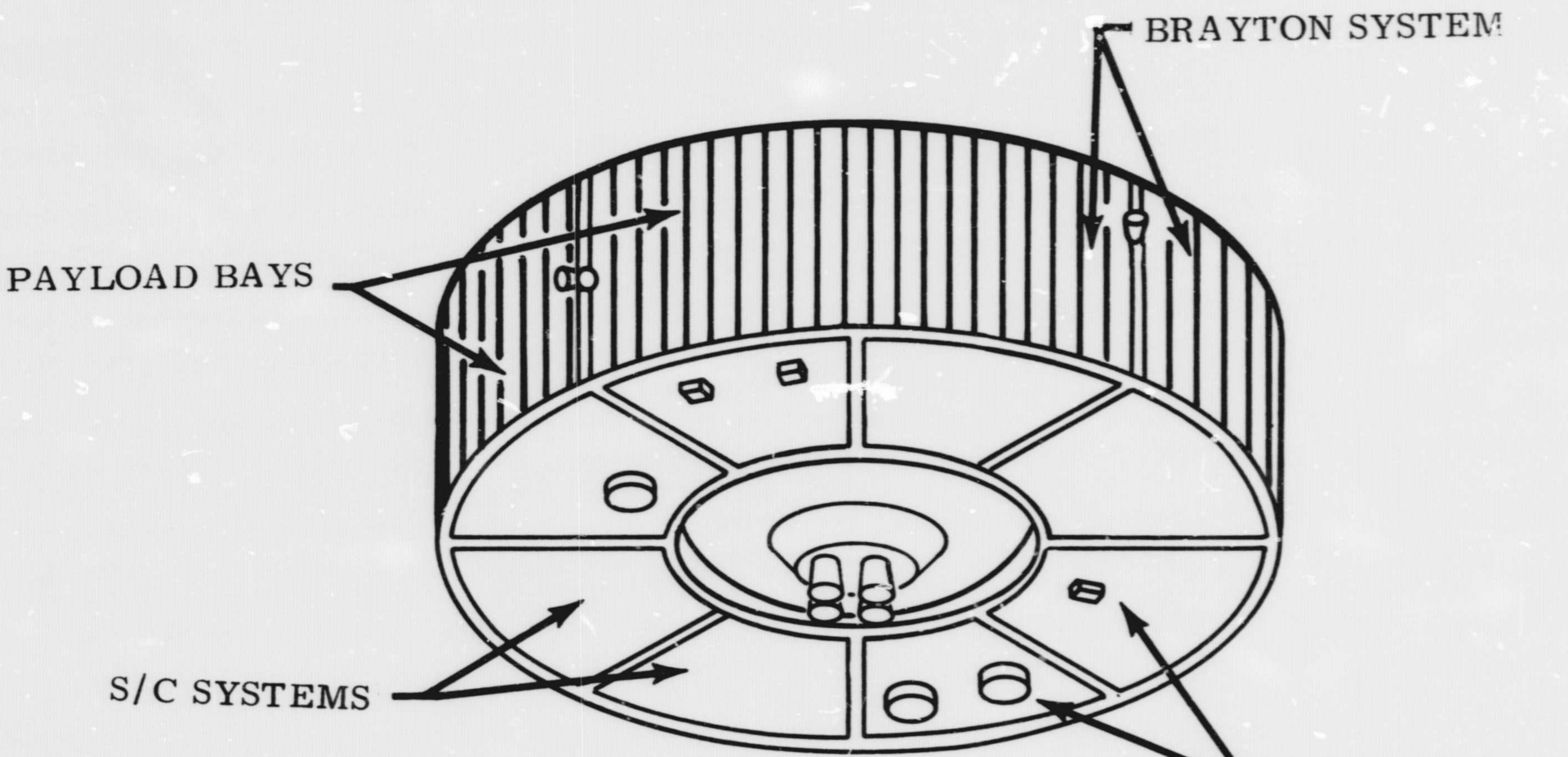


FIGURE S - 18 BRAYTON S/C

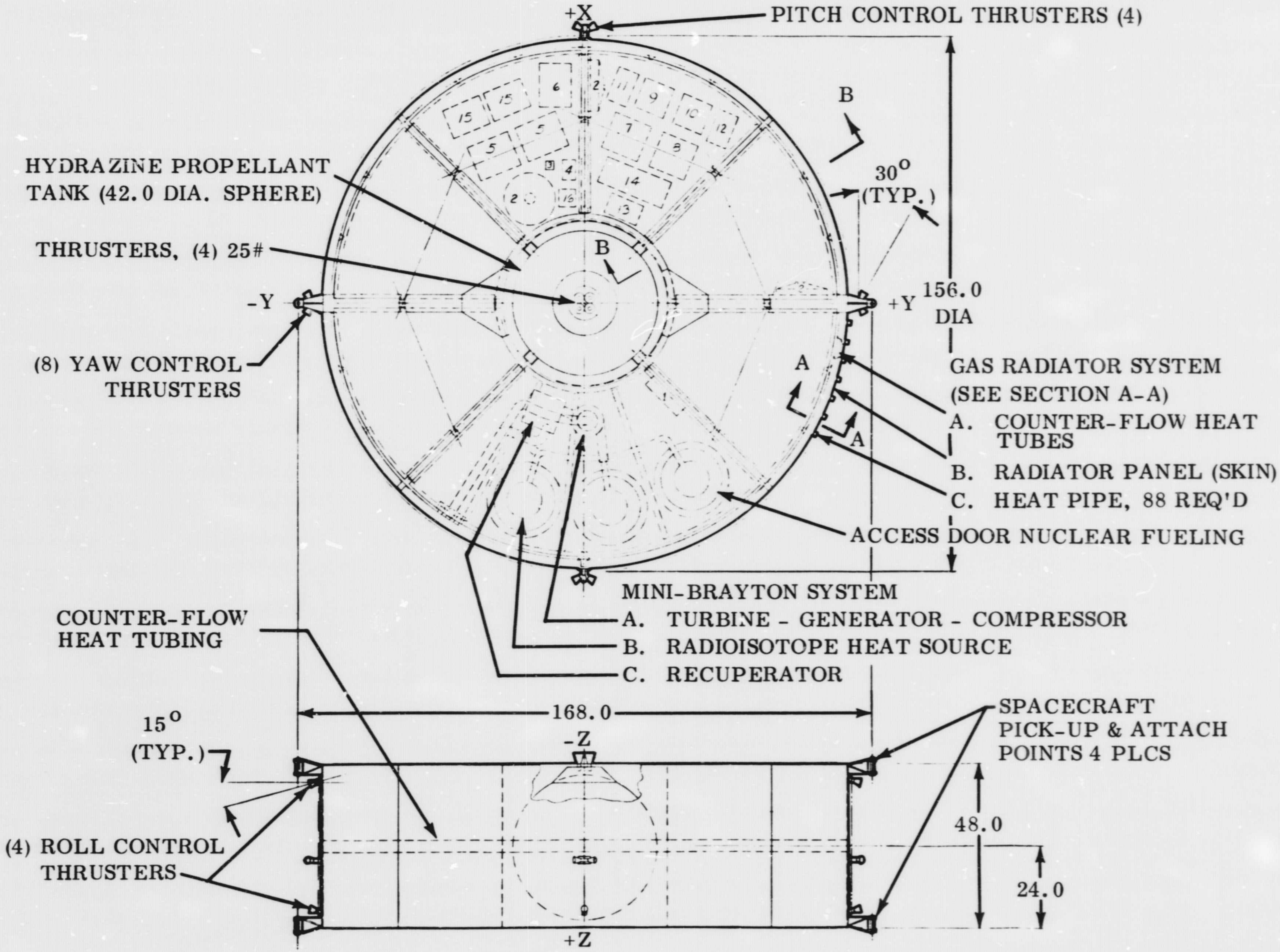
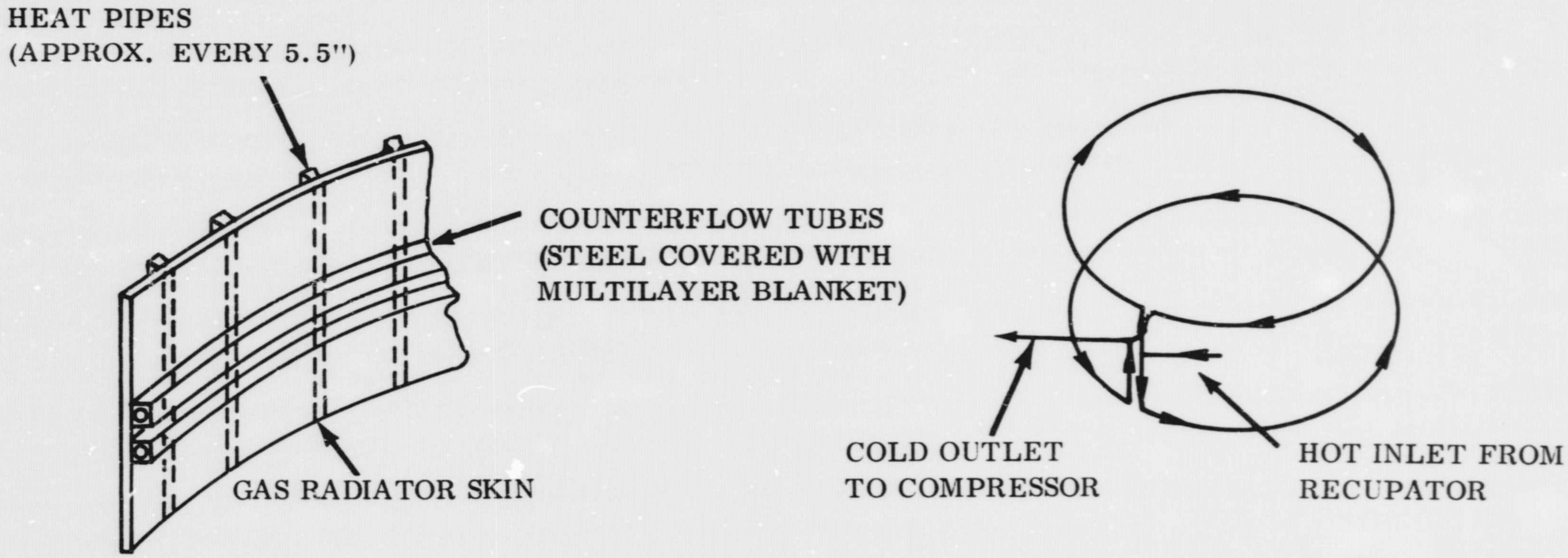


FIGURE S - 19 BASIC MINI-BRAYTON S/C



ISOTHERMAL GAS RADIATOR CONSTRUCTION

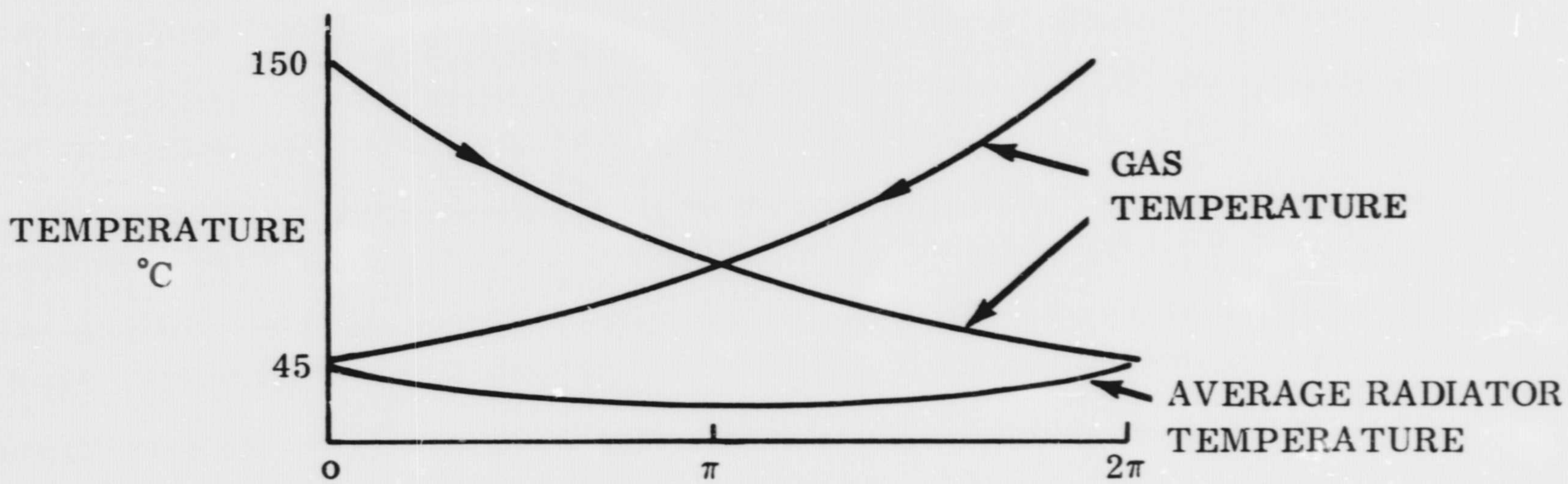


FIGURE S - 20 BRAYTON S/C RADIATOR

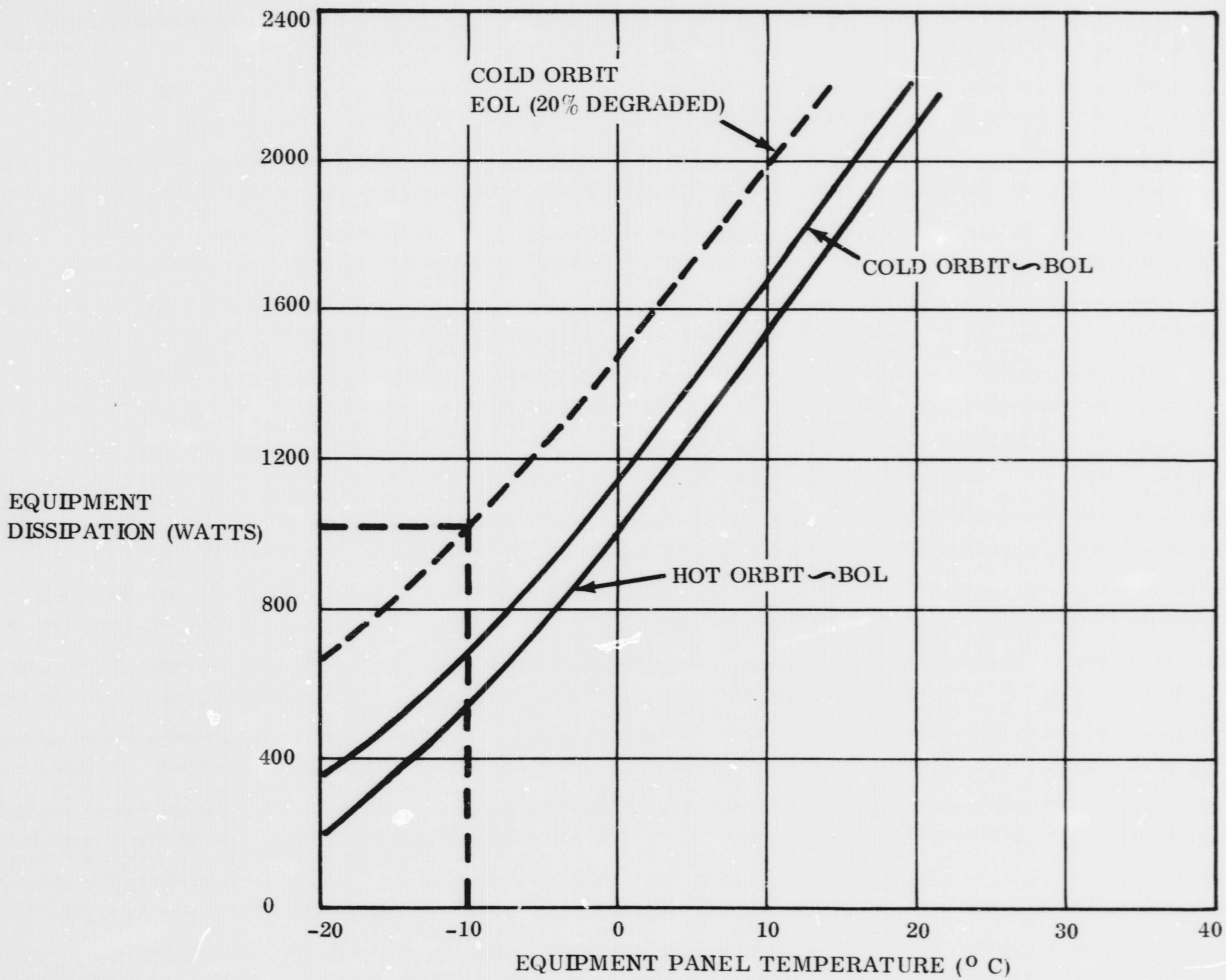


FIGURE S - 21 EQUIPMENT PANEL TEMPERATURES - BRAYTON S/C

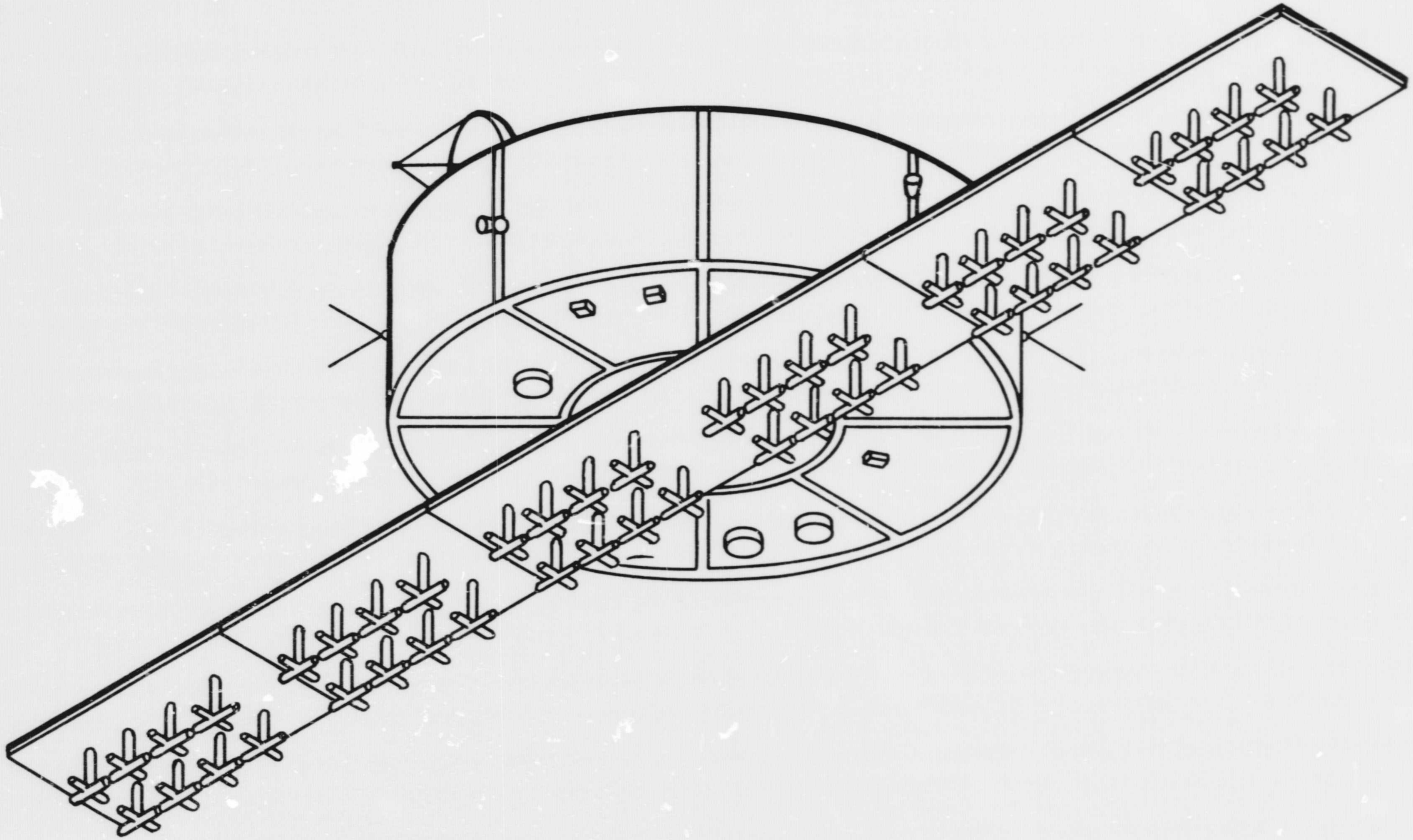


FIGURE S - 22 EOS MISSION ON BRAYTON S/C

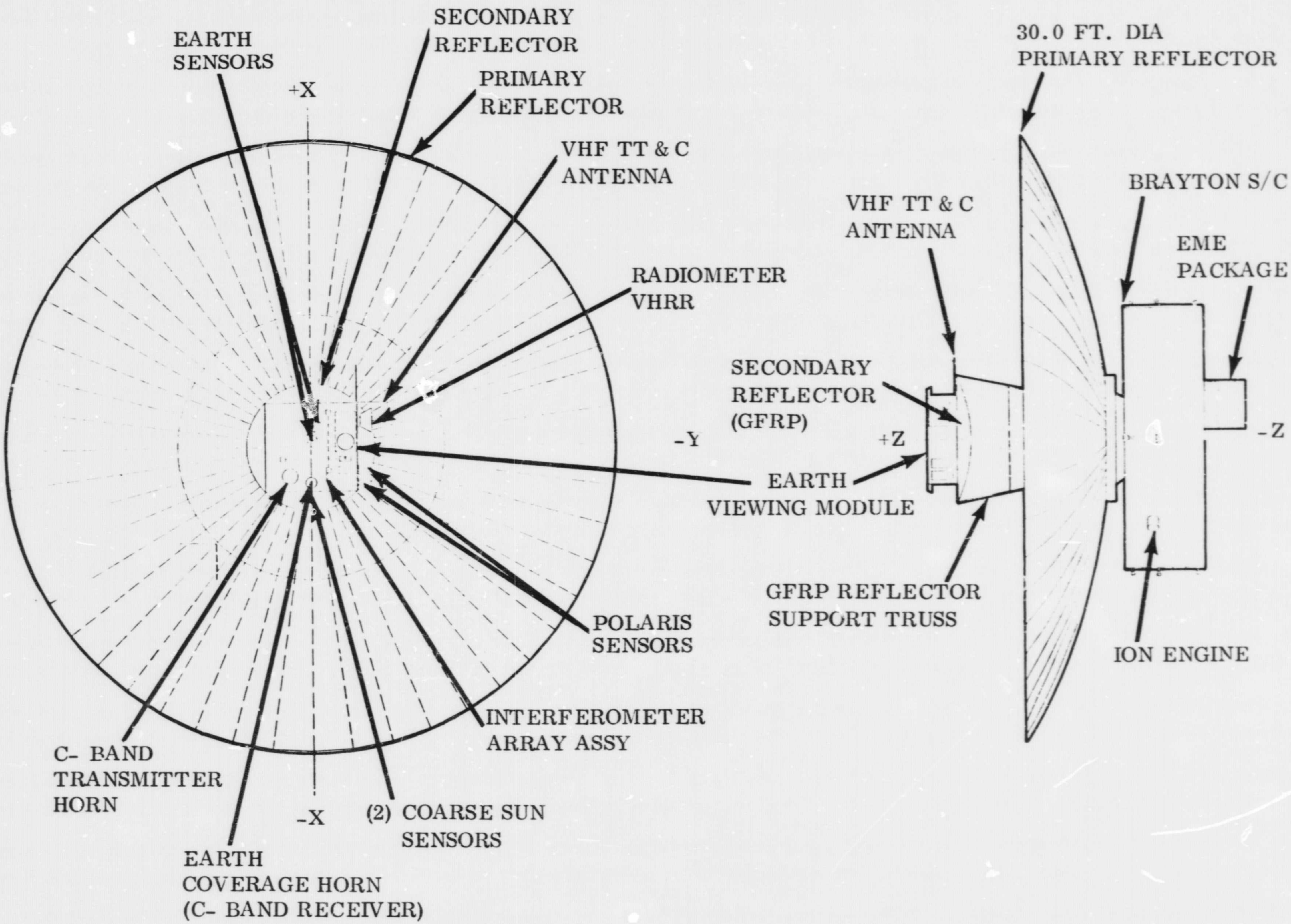


FIGURE S - 23 ATS-G MISSION DEPLOYED ON BRAYTON S/C

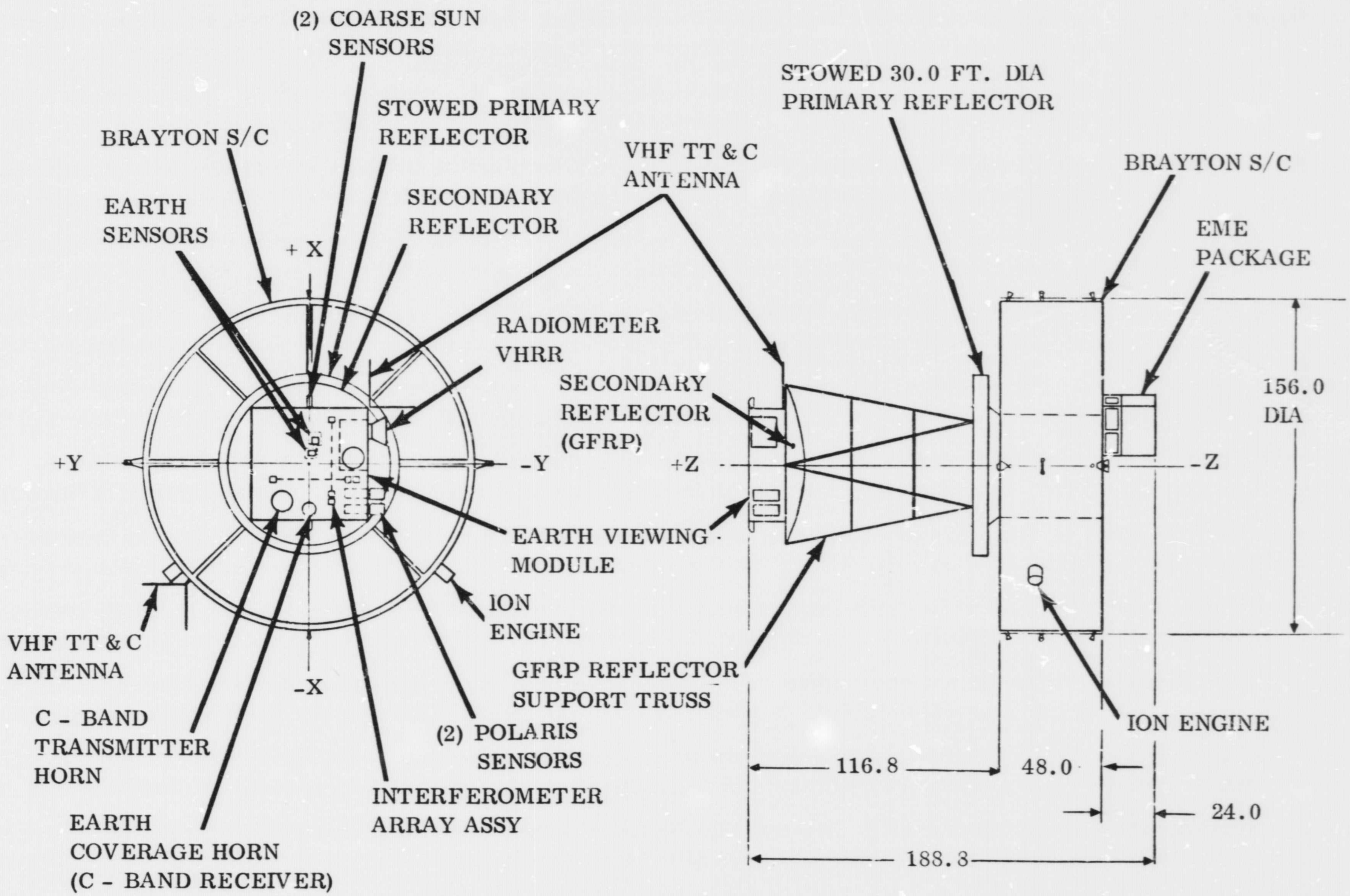


FIGURE S - 24 ATS-G MISSION STOWED ON BRAYTON S/C

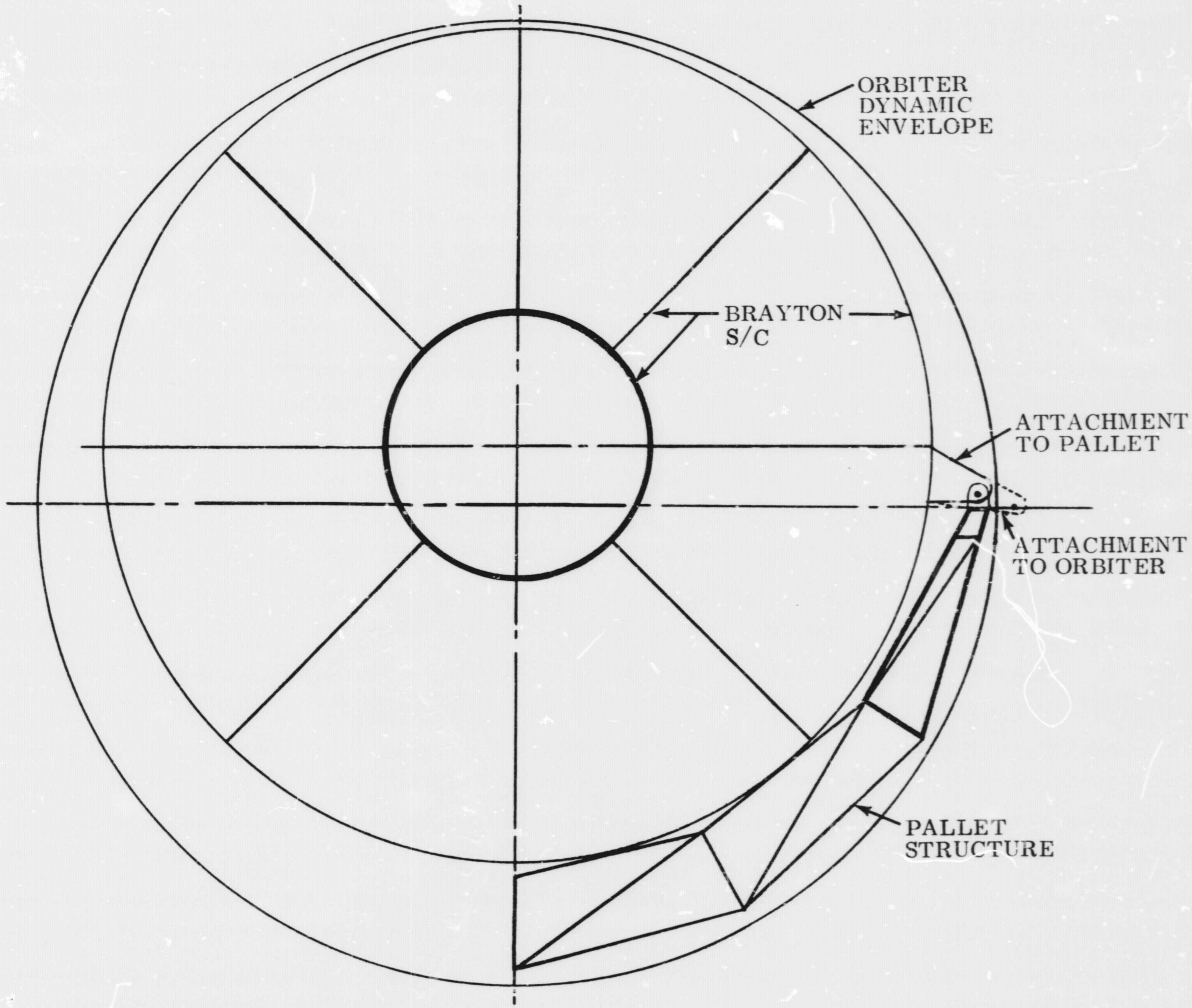


FIGURE S - 25 BRAYTON S/C INSTALLATION ON PALLET IN ORBITER

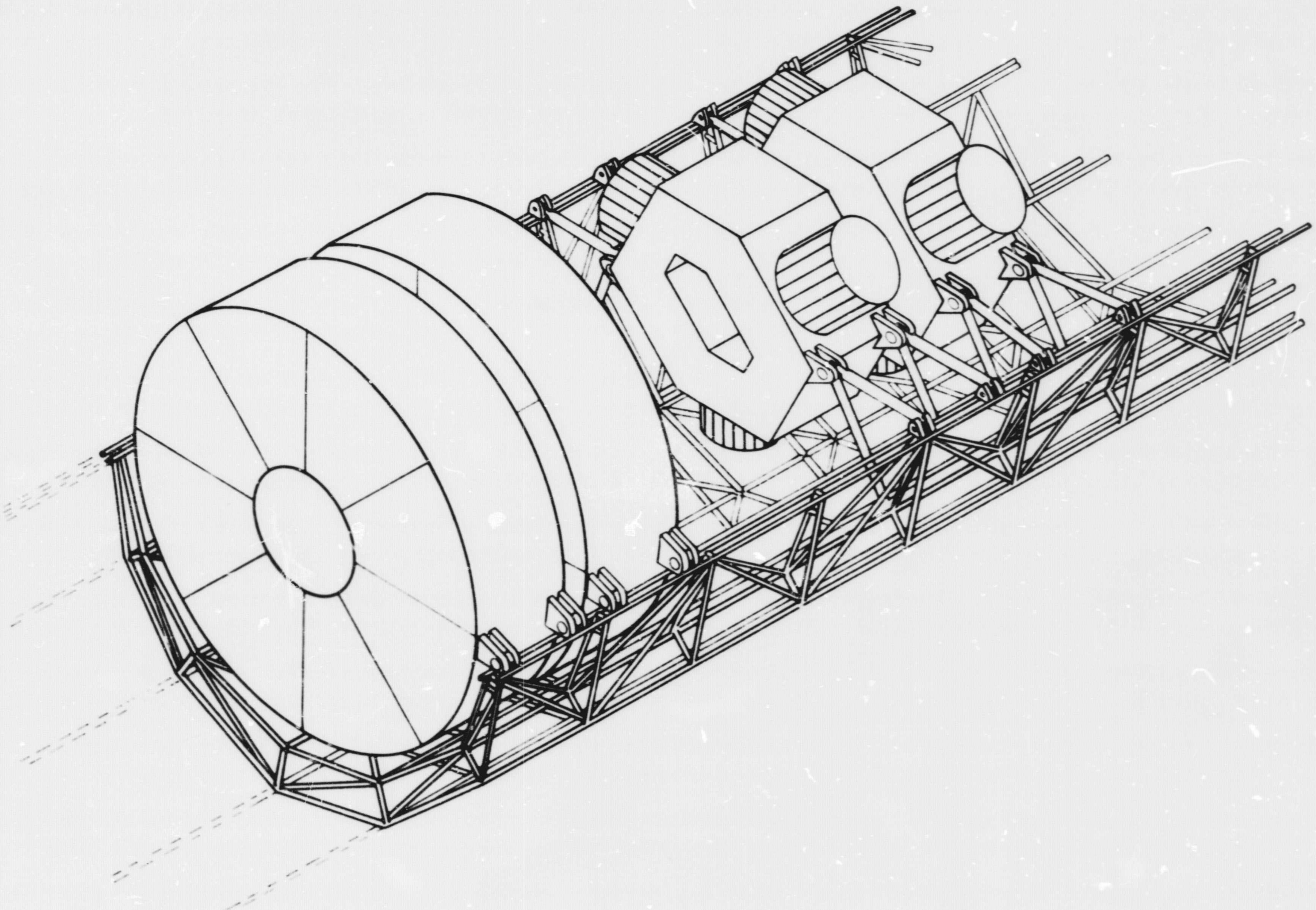


FIGURE S - 26 RTG AND BRAYTON S/C ON PALLET IN ORBITER

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